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The Relationship of Wind Correction Angle to Drift Angle

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SUMMARY

The relationship between correction angle and drift angle is treated mathematically and a formula for the relationship is derived. It is found that the correction angle is less than, equal to or greater than the drift angle according as the groundspeed (as observed when heading on the desired course) is, respectively, less than, equal to or greater than the airspeed.

INTRODUCTION

THE PURPOSE of this paper is to provide a definite and comprehensive answer to the long-standing question among aerial navigators concerning the exact nature of the relationship between wind correction angle and drift angle.

In the following discussion the term *drift angle* shall be used to mean the angle measured from the heading of an aircraft to its track when the heading is in the direction of a desired course. The term *wind correction angle* shall be used to mean the angle measured from the desired course to the heading necessary in order to make the track of the aircraft coincide with the desired course.

It is frequently assumed that the drift angle and wind correction angle are equal. While they may be equal under certain very special circumstances, and are approximately equal as long as the ratio of wind speed to airspeed is small (that is, they do not differ by much more than 1° as long as the ratio mentioned is less than 0.2), under other circumstances the difference between them may be considerable. This difference depends on the ratio of the speed of the wind to the true airspeed of the aircraft and also on the wind angle (which shall be used to mean the angle measured from the desired course of the aircraft to the direction from which the wind is blowing).

The following notations are used in this paper.

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a = the true airspeed of the aircraft
w = the speed of the wind
θ = the wind angle
D = the drift angle
C = the wind correction angle
r = w/a, the ratio of the wind speed to the true airspeed of the aircraft

Fig. 1 is a vector diagram illustrating the wind correction angle and the drift angle. Vector **OA** represents the true airspeed and heading of an aircraft in the direction of the desired course. Vector **AB** represents the velocity of the wind. The resultant vector **OB** thus represents the track and groundspeed of the aircraft and the angle **AOB** the drift angle. The vector **OA′** represents the true airspeed of the aircraft and the heading whose resultant with the wind velocity **A′B′** produces a track coincident with the desired course. Thus, vector **OB′** represents the track and groundspeed for such a heading and the angle **B′OA′** represents the correction angle.

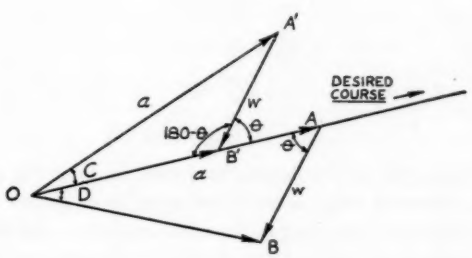

FIG. 1.

Using the Law of Sines in the triangle **OA′B′**:

$$(\sin C)/w = \sin (180° - \theta)/a$$

or: $(\sin C)/w = \sin \theta/a$, since $\sin (180° - \theta) = \sin \theta$

Hence: $\sin C = (w/a) \sin \theta$

or: $\sin C = r \sin \theta$

Using the Law of Sines in the triangle **OAB:**

$$(\sin D)/w = \sin \theta/\mathbf{OB}$$

and $$\sin D = w \sin \theta/\mathbf{OB}$$

Then, since $\mathbf{OB} = \sqrt{a^2 + w^2 - 2aw \cos \theta}$, by the Law of Cosines, $\sin D = w \sin \theta/\sqrt{a^2 + w^2 - 2aw \cos \theta}$ $= r \sin \theta/\sqrt{1 + r^2 - 2r \cos \theta}$

$$= \sin C/\sqrt{1 + r^2 - 2r \cos \theta}$$

whence: $\sin C = \sqrt{1 + r^2 - 2r \cos \theta} \sin D$

For C and D acute angles, $C \gtreqless D$ according as $\sin C \gtreqless \sin D$, and from the above equation it is seen that this will occur according as

$$\begin{aligned} \sqrt{1 + r^2 - 2r \cos \theta} &\gtreqless 1 \\ 1 + r^2 - 2r \cos \theta &\gtreqless 1 \\ r^2 - 2r \cos \theta &\gtreqless 0 \\ r(r - 2 \cos \theta) &\gtreqless 0 \end{aligned}$$

It is noticed that $C = D$ if $r = 0$. This is the no-wind case for which C and D are both zero. For the more general case where a wind is present, we have $C \gtreqless D$ according as

$$\begin{aligned} r - 2 \cos \theta &\gtreqless 0 \\ -2 \cos \theta &\gtreqless -r \\ -\cos \theta &\gtreqless -r/2 \\ \cos \theta &\lesseqgtr r/2 \\ \theta &\gtreqless \arccos r/2 \end{aligned}$$

Stating the above results in words, we have the rule: *The correction angle is less than, equal to or greater than the drift angle according as the wind angle is, respectively, less than, equal to or greater than the angle whose cosine is one-half the ratio of the wind speed to the true airspeed of the aircraft.*

As shown above, when heading in the direction of the desired course **OA**, Fig. 1, the groundspeed is:

$$\mathbf{OB} = \sqrt{a^2 + w^2 - 2aw \cos \theta} = a\sqrt{1 + r^2 - 2r \cos \theta}$$

and it is seen that $\mathbf{OB} \gtreqless a$ according as $\sqrt{1 + r^2 - 2r \cos \theta} \gtreqless 1$ and hence according as $\theta \gtreqless \arccos r/2$. Hence the above given rule may also be stated in the following manner: *The correction angle is less than, equal to or greater than the drift angle according as the groundspeed, as observed when heading on the desired course, is, respectively, less than, equal to or greater than the airspeed.*

To find the value of the difference $C - D$ for general values of r and θ it is first necessary to find:

$$\cos C = \sqrt{1 - \sin^2 C} = \sqrt{1 - r^2 \sin^2 \theta}$$

$$\cos D = \sqrt{1 - \sin^2 D} = (1 - r \cos \theta)/\sqrt{1 + r^2 - 2r \cos \theta}$$

Then $\sin (C - D) = \sin C \cos D - \cos C \sin D$

$$= \frac{r \sin \theta \left(1 - r \cos \theta - \sqrt{1 - r^2 \sin^2 \theta}\right)}{\sqrt{1 + r^2 - 2r \cos \theta}}$$

and $\cos (C - D) = \cos C \cos D + \sin C \sin D$

$$= \frac{r^2 \sin^2 \theta + (1 - r \cos \theta)\sqrt{1 - r^2 \sin^2 \theta}}{\sqrt{1 + r^2 - 2r \cos \theta}}$$

Using the formula for $\sin (C - D)$ the values of $C - D$ have been computed and are presented in Table 1 for values of θ at 10° intervals from 0° to 180° and for the values of r: 0.1, 0.2, 0.3, 0.4, 0.5, 0.75 and 1. The data given therein are also presented graphically. Figs. 2 and 3 consist of curves representing the formula

$$C - D = \arcsin \frac{r \sin \theta \left(1 - r \cos \theta - \sqrt{1 - r^2 \sin^2 \theta}\right)}{\sqrt{1 + r^2 - 2r \cos \theta}}$$

for the above given values of r.

Maximum Difference between Drift Angle and Correction Angle

The aspect of the study of the correction angle and drift angle problem which is of most interest to the practical aerial navigator lies in the determination of the wind angle for which the absolute value of the difference $C - D$ will be a maximum for any particular value of the ratio r, and hence the determination of the values of such maximum differences. This will enable him to know under what circumstances the drift angle may or may not be taken as a safe approximation to the correction angle. This information may be obtained with sufficient accuracy for most purposes from Tables 1 and 2, and the curves shown in Figs. 2 and 3. For a more precise solution of the problem the following method may be used.

Since $\cos (C - D)$ has extremuous values for the same values of θ as those for which $C - D$ itself has such values and also for those values of θ for which $C - D = 0$ it is possible to find the values of θ for which the absolute value of $C - D$ is a maximum by finding the values of θ for which $\cos (C - D)$ has maximum and minimum values and discarding those values of θ for which $C - D = 0$. The values of θ for which $\cos (C - D)$ has maximum or minimum values may be found by equating to zero the derivative with respect to θ of $\cos (C - D)$. Setting $d \cos (C - D)/d\theta$ equal to zero gives rise to the equation,

$$\sin \theta (2x - r)[3rx^4 + (-2r^2 - 4)x^3 + 3rx^2 + (-2r^2 + 2)x + r^3 - r] = 0,$$

TABLE 1

Values of $C - D$ for Various Wind Angles and Values of r

	Values of r						
θ	0.1	0.2	0.3	0.4	0.5	0.75	1
0°	0°00′	0°00′	0°00′	0°00′	0°00′	0°00′	(−90°00′)
10°	−0 06	−0 29	−1 15	−2 33	−4 44	−18 38	−75 00
20°	−0 11	−0 53	−2 15	−4 30	−8 01	−26 07	−60 00
30°	−0 15	−1 14	−2 50	−5 52	−9 19	−24 52	−45 00
40°	−0 16	−1 17	−2 56	−5 26	−8 46	−19 43	−30 00
50°	−0 16	−1 09	−2 39	−4 34	−6 56	−12 55	−15 00
60°	−0 14	−0 55	−1 57	−3 10	−4 20	− 5 31	0 00
70°	−0 10	−0 36	−1 05	−1 27	−1 31	+ 1 20	+15 00
80°	−0 04	−0 10	−0 08	+0 15	+1 10	7 13	30 00
90°	+0 02	+0 14	+0 45	1 47	3 26	12 12	45 00
100°	0 08	0 35	1 28	2 58	5 08	14 25	60 00
110°	0 12	0 51	2 01	3 47	6 09	15 32	75 00
120°	0 16	1 02	2 19	4 10	6 32	15 13	90 00
130°	0 18	1 05	2 21	4 09	6 21	13 52	75 00
140°	0 18	1 02	2 12	3 46	5 40	12 06	60 00
150°	0 15	0 52	1 50	3 05	4 34	9 14	45 00
160°	0 11	0 38	1 19	2 11	3 11	6 16	30 00
170°	0 06	0 20	0 41	1 08	1 39	3 12	15 00
180°	0 00	0 00	0 00	0 00	0 00	0 00	00 00

TABLE 2

Values of θ for Which $C - D = 0$ (and Hence $C = D$)

r	0.1	0.2	0.3	0.4	0.5	0.75	1
θ	87°08′	84°16′	81°22′	78°28′	75°31′	67°59′	60°00′

where cos θ is represented by x. The factors sin θ and $(2x - r)$ when equated to zero give the values $\theta = 0°$, $\theta =$ arc cos $r/2$ and $\theta = 180°$, for each of which cos $(C - D)$ is a maximum but for which $C - D = 0$. Hence these roots are disregarded and the pertinent roots are found as the two roots, lying, respectively, between -1 and 0 and between 0 and 1, of the residual equation,

$$3rx^4 + (-2r^2 - 4)x^3 + 3rx^2 + (-2r^2 + 2)x + r^3 - r = 0$$

The four roots of this equation are:

(Where the double signs occur, the positive or negative sign is to be taken according as the quantity $-(r^3/27) - (5r/9) + (8/9r) - (8/27r^3)$ is positive or negative.)

$$x_1 = (r/6) + (1/3r) \pm (-\sqrt{t_1} - \sqrt{t_2} - \sqrt{t_3})$$

$$x_2 = (r/6) + (1/3r) \pm (-\sqrt{t_1} + \sqrt{t_2} + \sqrt{t_3})$$

$$x_3 = (r/6) + (1/3r) \pm (\sqrt{t_1} - \sqrt{t_2} + \sqrt{t_3})$$

$$x_4 = (r/6) + (1/3r) \pm (\sqrt{t_1} + \sqrt{t_2} - \sqrt{t_3})$$

where: $t_1 = (r^2/36) - (1/18) + (1/9r^2) + u + v$

$$t_2 = (r^2/36) - (1/18) + (1/9r^2) + eu + e^2v$$

$$t_3 = (r^2/36) - (1/18) + (1/9r^2) + e^2u + ev$$

$$e = (-1 + \sqrt{-3})/2$$

$$u = \sqrt[3]{P + \sqrt{Q}}, v = \sqrt[3]{P - \sqrt{Q}}$$

$$P = (1/2^6 3^3)[2r^4 - 2r^2 - 1 + (2/r^2)]$$

$$Q = (1/2^{10} 3^9)[27r^8 - 182r^6 + 624r^4 - 1317r^2 + 1750 - (1425/r^2) + (651/r^4) - (128/r^6)]$$

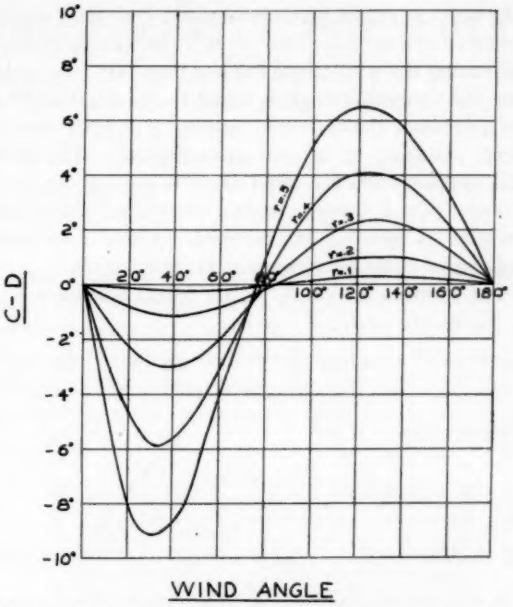

FIG. 2. Values of the difference $C - D$ for all wind angles and for certain values of r.

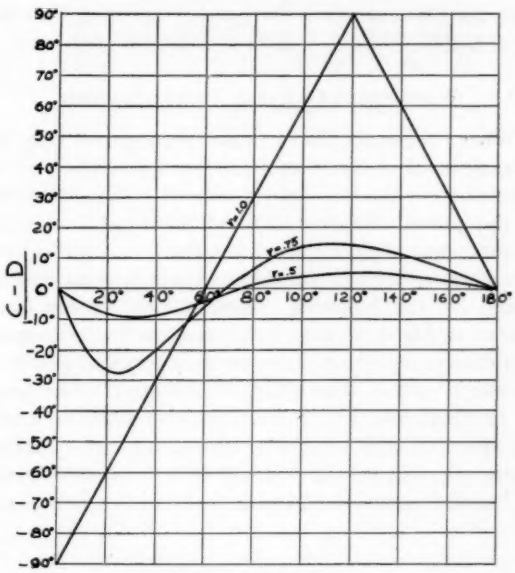

FIG. 3. Values of the difference $C - D$ for all wind angles and for certain values of r.

From the nature of the problem it is obvious that two of the values of x, that is, of cos θ, will be real and will lie between -1 and 1 while the other two must either be imaginary or lie outside this range of values.

WIND SPEED AND AIRSPEED EQUAL

It is interesting to note the special results which occur for the value $r = 1$, when the wind speed is equal to the true airspeed of the aircraft. The relation-

ship sin $C = r \sin \theta$ then becomes sin $C = \sin \theta$ whence either $C = \theta$ or $C = 180° - \theta$. The former case applies when the wind angle θ is less than 90°. In such a case the correction angle is equal to the wind angle itself and hence the corrected heading is directly into the wind, resulting in a zero groundspeed. The latter case applies when the wind angle is greater than 90°. *Hence a pilot desiring to make a down-wind course when the wind is equal to his airspeed, applies a correction angle equal to the supplement of his wind angle.*

The formula for sin $(C - D)$ reduces, when $r = 1$, to the simple form:

$$\sin (C - D) = -\cos (3/2)\theta = -\sin [90° - (3/2)\theta] = \sin [(3/2)\theta - 90°]$$

whence either

$$C - D = (3/2)\theta - 90°$$

or

$$C - D = 180° - [(3/2)\theta - 90°] = 270° - (3/2)\theta$$

The first value of $C - D$ is to be taken for values of θ between 0° and 120° and the second is to be used for values of θ between 120° and 180°. As θ approaches zero for this case (and the ground speed approaches zero) the difference $C - D$ approaches 90° as a limit.

Wind Speed Greater Than Airspeed

When r is greater than 1 (wind speed greater than the true airspeed of the aircraft) correction angle is of no interest since the track of the aircraft is then restricted in the general direction in which the wind is blowing. The greatest angle by which the track may differ from the direction toward which the wind is blowing may be found by the following considerations. From **O** as center, in Fig. 4, a circle of radius equal to the airspeed a is drawn, representing the locus of terminal points of airspeed vectors drawn for all headings. The locus of terminal points of a constant wind velocity vector (of speed w) added to such airspeed vectors is another circle of radius a with center at **O′** a distance w from **O** in the direction toward which the wind is blowing, and

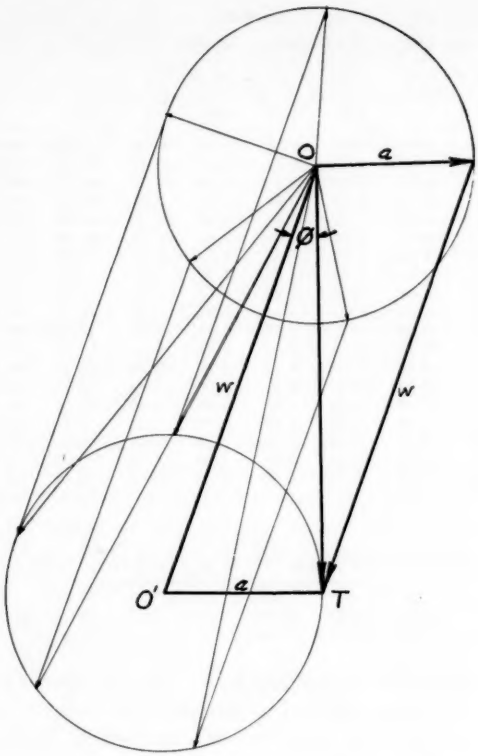

Fig. 4.

hence the vectors representing the track and groundspeed of the aircraft are drawn from **O** to points on this second circle. The maximum angle ϕ by which the resultant track of the aircraft may diverge from the direction toward which the wind is blowing will be obtained when the track is tangent to the second circle as illustrated by vector **OT** in Fig. 4. Since the tangent **OT** is perpendicular to the radius **O′T**, **OO′T** is a right triangle and it is then seen that sin $\phi = a/w$ whence the maximum angle of divergence is:

$$\phi = \text{arc sin } (a/w) = \text{arc sin } (1/r)$$

The correction angle to be applied in this case is 90°.

Development Problems of Light Aircraft Engines

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Aircooled Motors Corporation

EARLY IN 1938 it became apparent to the Aircooled Motors Corporation that the trend of the power requirements for the smaller airplanes would be constantly toward higher limits. A long range engineering program was undertaken with the idea of eventually having a series of engines in production at an economical cost with a power range from 65 to possibly 400 hp.

As yet the sales volume of the light airplane is such that it does not permit large expenditures for mass production of the airplane or engine. For this reason, the engine company is faced with the problem of making their designs flexible enough so that several models can be produced on a basic set of tools. This fundamental has been followed in the design of the power plants mentioned above. The importance of volume production is shown in Stout's statement that the airplane field today needs a 100 hp. engine weighing 100 lbs. for $100 at a production of 100 a day. It is felt that this is possible to meet, providing there is a volume of 100 a day and when the $100 refers to the labor and material bill of the engine maker and not to the selling price of the engine.

After arriving at the definite sizes for the projected series of engines, further studies were made to determine type or shape of these engines. Considering various types from all angles it was concluded that the horizontal opposed type of from four to twelve cylinders would work out to best advantage for our particular type of plant equipment.

Briefly, this decision was based on the fact that the Aircooled Motors Corporation had had experience in building the horizontal opposed type on their 50 hp. engine, the power plant was compact, economical to build and very simple from a lubrication standpoint, inasmuch as dry sumps, etc., would be eliminated. Furthermore, the horizontal opposed type could also be readily converted for applications other than aircraft by the addition of cooling fan, air housing, flywheel housing, etc.

The first consideration in any engine after the particular type and size is determined, is the crankcase. Various cases were designed—that is, the barrel type of case in which the shaft would enter from one end, the horizontal split type of case where the separation passes through the center of the cylinder pilot holes, and the final solution was to make the case in two halves, splitting it vertically through the crankcase axis. This type permitted the use of a separate oil sump which could be easily attached and varied according to the particular application. Also, in it was incorporated a separate crankcase cover which would permit various types of mountings, accessories, etc. (See Fig. 1.)

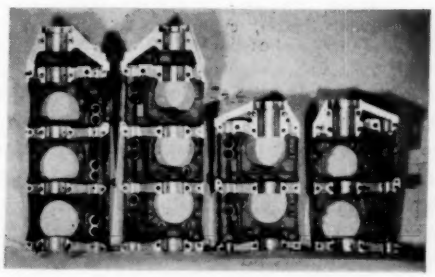

FIG. 1. Finished crankcases, 4 cylinder and 6 cylinder engines.

The basic engine in this new series of engines is the 4AC-176, 65 hp. engine of four cylinders, 4″ bore and $3\frac{1}{2}$″ stroke, introduced in March, 1940. This engine was followed immediately with the 80 hp. type, which is basically the same engine except the compression ratio is increased from 6.3 to 7.1 and the engine speed increased to 2500 r.p.m. where 80 hp. is developed.

The first engine incorporated a crankcase of high strength iron alloy having $\frac{1}{8}$″ walls. The weight of the crankcase was very satisfactory. Nevertheless, we found it practically impossible to obtain uniform castings without minute foundry cracks. Consequently, the iron alloy was eliminated as a material and the design changed to incorporate an aluminum alloy, feeling that perhaps later on, when more time was available and more customer pressure for lower cost engines, the cast iron alloy case could again be revived.

The decision on the particular type of cylinder to use on the new series of engines required an extreme amount of consideration as it was concluded that there were any number of designs which would operate satisfactorily, such as an all-aluminum cylinder with a thin iron liner, a steel barrel with machined fins screwed into an aluminum head, or finally, an iron alloy casting screwed into an aluminum head.

This last construction was the one chosen. The reasons for this choice were the fact that the Aircooled Motors Corporation had already had in production a

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commerical engine which had proved to be economical to manufacture, and it was felt that the service problems would be much less than the other two types mentioned, principally from a piston ring or oil consumption standpoint. Of course, it is admitted that this type of construction is slightly heavier than the other two mentioned. Nevertheless, from a cost standpoint it is possible to concentrate on other parts of the engine for reduced weight at a decided overall saving in the cost of the engine over using the expensive first two types mentioned. (See Fig. 2.)

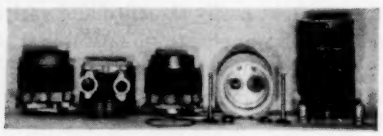

FIG. 2. Cylinder—assembly parts (rough cylinder barrel, cylinder head castings, also machined castings).

One advantage from a service standpoint which this construction has is the fact that the cylinder barrel can be removed from the cylinder head by heating the assembly up to 600 degrees Fahrenheit and unscrewing the barrel. The barrel can either be replaced or a new barrel installed in the head, inasmuch as the joint between the cylinder head and the barrel is sealed three ways—that is, the gasket between the head and the barrel, the shrink fit on the land above the head and the shrink fit on the land below the threads.

In the design of the crankshaft, similar practice has been followed on the 4–6 and 8 cylinder engines—that is, the 4 cylinder engine uses a four-throw shaft, the 6 cylinder engine a six-throw shaft and the 8 cylinder engine an eight-throw shaft, whereas the 12 cylinder engine uses a conventional three-throw shaft with two connecting rods on a given crank pin and with a main bearing between each crank pin. No doubt the main bearing between each crank pin is an advantage. However, it is not possible to do this on the 4–6 and 8 cylinder types without making the engines extremely long. This is out of the question on the entire group of engines, as all the engines are built on the same cylinder centers; that is, 6″ and of course, have identically the same cylinder assemblies excepting the engines with the smaller or 4″ bore.

All the engines, of course, have overhead valves operated through push rods. The clearance is automatically taken up by the Wilcox-Rich hydraulic valve lifters. The lifters have been standard practice in all Franklin engines built since 1933—in fact the Franklin engine is the second American engine to use this type of lifter. The experience gained in the many years from the use of this unit has resulted in a background which is invaluable in the development of a new engine. The construction of the Wilcox-Rich hydraulic lifter is well understood. However, the proper functioning of this unit requires careful cooperation between the engineers of the Wilcox-Rich Division, also the valve spring maker and valve maker and the engine manufacturer, inasmuch as there are many problems in the valve gear train which require utmost care in design.

The intake systems for the entire group of engines are basically the same; that is, the carburetor is mounted on the oil pan with the individual pipes from each cylinder passing to the union at the pan and hence to the carbureter. This construction has been followed consistently, inasmuch as it is simple and gives very fine results. The intake manifolds themselves were laid out as a group by the late Clarence Kirkham, Chief Engineer of the Swan Carbureter Company, and his principles followed as closely as possible. Experience has shown that manifolds following the Swan patents result in increased power and economy and at a minimum development cost.

The entire series of engines has been designed with the idea of keeping the overall height to a minimum—in fact it is possible to use dry sump lubrication eliminating the oil pan and maintaining a maximum height of 13″. When this practice is followed the carbureters are located at the rear of the engine. Thus far no active work has been done in the dry sump types of engines except for a few experimental installations.

For lubrication a conventional system is used; that is, the oil pump located on the bottom of the engine with an extension into the oil sump. It has been the practice of the Aircooled Motors Corporation to use a pump with very large capacity with a relief valve set for 35 to 45 lbs., and they found that for normal type of operation the relief valve is operating, thus maintaining a factor of safety of supply to take care of excessive clearance conditions after an engine has been in service for a great length of time. The pump furnishes oil to the main bearings, connecting rod bearings, hydraulic valve lifters, valve mechanism, by direct pressure. Other internal parts of the engine are lubricated by splash.

The bearings of the engine are in the main precision interchangeable steel back copper lead faced, except the camshaft bearings which are interchangeable precision steel back babbitt faced. For the 4–6 and 8 cylinder engines the majority of the bearings are interchangeable. The 12 cylinder engine has larger bearings. The 4–6 and 8 cylinder engines incorporate connecting rod bearings $1^{15}/_{16}$″ in diameter, main bearings $2^{1}/_{4}$″ in diameter and upper connecting rod bushings 1″ in diameter. The 12 cylinder engine has $2^{1}/_{4}$″ diameter crank pins, $2^{3}/_{4}$″ main bearings and the same size upper piston pin bearings as all the other engines in this series.

For piston design the engines use specially developed aluminum alloy pistons with a steel strut which permits the pistons to be fitted in the 4″ bore and $4^{1}/_{4}$″ bore with approximately .003″ clearance, which is quite unusual for aircraft engines. The pistons were selected

with the idea of eliminating all piston slap, also to assist in oil control. The same identical piston is used in the entire series of engines for a given bore.

For electrical equipment the engines use conventional Eisemann magnetos and Auto-Lite revised automotive type of equipment. This applies to the 4–6 and 8 cylinder engines. However, on the 12 cylinder engines we will offer either Auto-Lite automotive type of equipment or the specialized aircraft equipment. It is felt that this 12 cylinder engine with the higher output and more expensive applications warrants the use of the more costly electrical equipment.

The above covers the general construction of the entire group of engines. However, it is felt that it will be necessary to mention specifically certain problems arising in each engine in the development program.

Mention was made previously of the trouble encountered in the attempt to use a crankcase of cast iron in the 4 cylinder engines. It was never the intention to use iron in the other engines excepting the unit with the lowest power, which, of course, sells at the lowest price. The use of the aluminum alloys in the other engines has followed the conventional practice with continual improvement in the power output of these models during the development program it has been found necessary to add more metal in the crankcase in order to stand up under continuous loads. Furthermore, it might be mentioned that it is very difficult at the present time to obtain castings in the desired alloys with delivery dates at all which are satisfactory. Consequently, it was found necessary to use alternate alloys and in certain cases increase the sections to permit their use.

In general it is felt that the major development problems encountered are confined to the cylinder assembly itself, as the problems encountered within the crankcase are those which are usually present in any engine development. Therefore, it is felt that it might be well to mention some of the problems in the cylinder.

In the first 4 cylinder engine of course an attempt was made to keep the weight of the cylinder barrel to a minimum. For that reason after approximately 100 hours of testing it was found that the cylinder flange at times would be cracked. Investigation showed that this trouble was caused by two things: first, the flange was, no doubt, too thin and improperly ribbed; second, a gasket of too great thickness was used, as when the cylinder was pulled down the ears on the flange were distorted, resulting in a fracture. The trouble was entirely eliminated by strengthening the flange and using a gasket approximately 0.007″ thick.

Another problem encountered in the engine was cracking of the cylinder head at the threads. Whereas the commercial engine using this type of construction, ran a brake mean effective pressure of not over 105 lbs., this new series of engines was operating as high as 150 lbs. For that reason it was apparent that manufacturing limits which were acceptable in the lower output engines would not work at all for the higher outputs. A study of the threads indicated that a sharp point was present at the root of the threads, resulting in extreme stress concentration. This trouble was eliminated by providing a proper radius at the root of the thread, also maintaining the proper shrink on the lands above the thread so that the heat of the explosion would not reach the upper thread in the aluminum, which is very weak, of course, under high temperatures.

Valve seats which would operate satisfactorily in engines up to 130 lbs. brake mean effective pressure were found to fail in an engine with a pressure of 145 lbs. Other aircraft engines and commercial engines, built by the Aircooled Motors Corporation, used niresist exhaust valve seats; that is—an iron alloy of 21 per cent nickel and 14 per cent chrome. After 100 hrs. in this new series of engines it was found that the seats would be pounded very badly, resulting in excessive leakage. Experiments with bronze seats found this material to be lacking as a satisfactory material. The problem was finally solved by the use of a stainless steel exhaust seat. The intake seat, however, never caused any trouble, being of ordinary conventional automotive cast material, designated as "moly" iron.

For valve guides a material of iron alloy was used for both the intake and exhaust. These guides are hardened and honed to the required size prior to assembly in the cylinder. After assembly in the head it is found that the bore of the guide is "pulled in." When the engine is hot, the bores are straight. We find this superior to honing the guides after assembly in the heads. It is not felt that the iron alloy guides can be used for fuels above 80 octane, inasmuch as experience has shown us that as the lead content of the fuel increases, corrosion results in the guide, causing excessive clearance. In other words, for fuels above 80 octane a bronze alloy exhaust guide would be needed. Experience shows that the intake guide in iron alloy is satisfactory.

Next to the problems encountered in the cylinder assembly the valve mechanism has required the greatest attention. Mention was made previously of the importance of close cooperation between the maker of the hydraulic lifter and the engine manufacturer. Fig. 3 shows the condition of improper valve action, resulting from the hydraulic valve lifter where the valve spring is selected improperly, causing a pump-up condition. It is noted that the power curve shows a decided irregularity. This is caused by a slight period in the spring at that point. This trouble was corrected by the use of a spring with a slight change in frequency. (See also this same curve for the results after the proper spring was installed.)

A maximum speed of 3100 r.p.m. was selected when the original line of engines was worked out. However, present experience indicates the advisability of operating these engines at 3500 to 3600 r.p.m. for take-off. This means that after take-off the engine is very likely

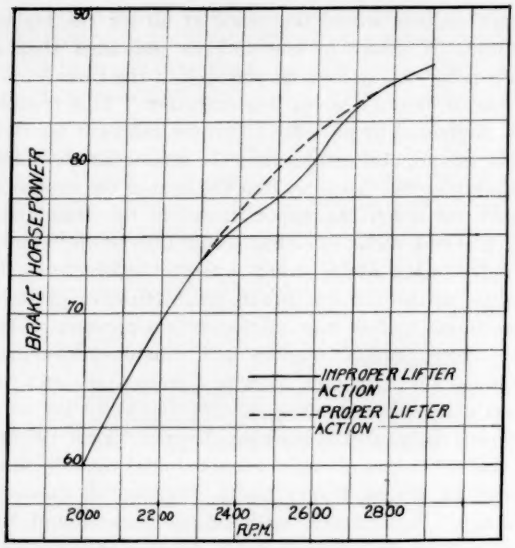

FIG. 3. Horsepower curve with improper valve lifter action, also with proper lifter action.

to momentarily speed up to 4000 r.p.m. To take care of this possibility it has been necessary to study the valve mechanism so that the engine would operate satisfactorily at the high speed. Whereas the valve gear of the engine would operate satisfactorily at 3100 r.p.m. it was found that it is quite a difficult task to get it to operate much above that without certain changes in the valve and cam mechanism. Investigation showed that at certain speeds the camshaft would deflect enough on one side of the engine to open valves on the other side. The use of a camshaft with a heavier diameter between bearings helped greatly on this score. Nevertheless, the problem was not completely solved by this arrangement. It was found that it was definitely necessary to use intermittent registration—that is, admit oil to the hydraulic lifter only when the valve was off the seat. This necessitated a close watch in the machining of the oil channel feeding the hydraulic lifters, as it was very easy for the drill to wander enough to go from intermittent registration to positive registration.

The final solution to the problem was the use of a camshaft with heavier diameter, the use of single valve springs $^3/_8$" longer than our regular type incorporating a friction type of dampener in the lower coils. With this construction it is possible to operate the engine up to 4000 r.p.m. without any indication of erratic valve operation.

Speeds of 3500 r.p.m. for take-off, of course, are very unusual, inasmuch as one would normally feel that the engine for cruising would be turning in the neighborhood of 3000 r.p.m. Nevertheless, in this development a gear reduction is used which is in gear for take-off and in direct drive for cruising. In other words, with the engine turning 3500 r.p.m. for take-off, the propeller turns 2450 r.p.m., using a .7 ration in the gear box. The engine cruises at approximately 2350 r.p.m. By "engine" the author refers to any one of the series of engines, inasmuch as they are all designed to incorporate the gear reduction unit. The problem encountered with this reduction unit is to have the maximum power for take-off at the same time without necessitating the engine to actually fly full throttle for cruising in direct drive. This necessitated a compromise in the power curve—that is, by taking less power at peak with an increase in power or torque at the 2300 to 2500 r.p.m. range. This has been accomplished with a minimum effect on the peak output, by advancing the cam timing 4 degrees and the use of dual carbureters, which increases the torque in the lower speed range by approximately 7 per cent. (See curve sheet Fig. 4.)

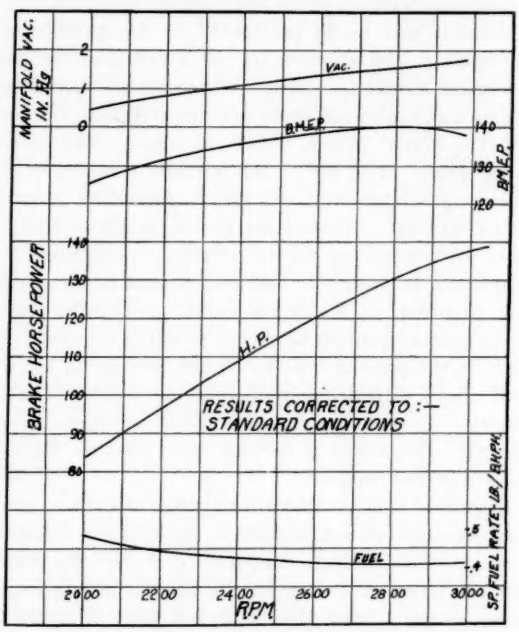

FIG. 4. Power curve with shift type gear reduction unit.

Of course, in a 6 cylinder engine using a single carbureter there is a decided interference between the suction impulses, which is entirely eliminated by the use of two carbureters with dual manifolds. For example: on the 6 cylinder engine with a firing order of 1–4–5–2–3–6, it is possible to absolutely separate the manifolds, having one manifold take care of one bank of cylinders and the other manifold take care of the other bank. These are cast into the oil pan and externally appear as if the carbureters were attached to the same manifold.

For the best all-around results the 6 cylinder engine uses two carbureters, the 8 cylinder engine 4 carbureters and the 12 cylinder engine four carbureters. At the present time there is no low cost dual up-draft aircraft carbureter in production. For this reason it is necessary to go along, for the time being at least, with the separate units. These remarks pertain to naturally

aspirated engines. However, when these engines are supercharged, of course one carbureter only would be used.

Normally it is expected that the accessories on an engine require very little attention in the development program, as they are considered to be a more or less standard design. However, in their approach to the accessory problem, as mentioned previously, the Aircooled Motors Corporation decided to use revised automotive equipment, inasmuch as it could be purchased at a much more attractive price. However, it was realized at the time this decision was made it would require very close cooperation with the accessory manufacturers in order to provide satisfactory units; we also realized that the manufacturers of automotive electrical equipment were in large production and would resist any basic changes in the design affecting the tooling. For that reason selection of the particular type of units was made with the idea of taking units which would require the minimum of alterations. After looking over the various units in big production, it was decided to concentrate on the Auto-Lite units. The original changes in these units consisted chiefly in the use of aluminum end housings, both on the starter and generator, with the rest of the unit remaining absolutely standard. Early tests indicated that it was impossible to keep the oil out of the starting motor. Consequently, it was necessary to incorporate a spring type of oil seal. Oil trouble was also encountered in the generator. Therefore, a positive seal was required with an oil slinger. These changes eliminated the oil problem of the units.

However, the problem of driving the generator was one which required many months of effort and at times it seemed hopeless. It was found that in anywhere from 10 to 100 hrs., the teeth on the composition drive gear would shear off. The calculated loads on the teeth did not show excessive loading. Nevertheless, the teeth continued to fail. The use of an all steel drive gear was found to be no help, as after approximately 20 hrs., the armature shaft itself failed. This indicated that the trouble perhaps was caused by a slight torsional vibration in the crankshaft coming in resonance of the natural period or of a harmonic of the armature shaft. The introduction of a rubber flexible drive in this gear absolutely eliminated the trouble. In other words, the gear was so mounted through synthetic rubber grommets that it could be moved a few thousandths concentric with the axis of the shaft enough to dampen out the vibration.

All the manufacturer's tests and official C.A.A. tests on the engines were run with 6 volt starter and generator equipment with absolutely no trouble in starting, even though the engine and battery were very cold. However, when installed in an airplane, it was found that cold starting was not a problem, but hot starting a major problem. In other words, after a ship taxied around the field with very little air on the engine and the engine stopped, it was difficult for the starter to crank it. First off it appeared that the answer would be to mount the battery next to the starting motor, decreasing the voltage drop through the system. This was not found to be the answer. Consequently, experiments were made with an 8 volt battery, next a 10 and finally a 12. The 12 volt battery was found to give positive cranking, even though the engine was so hot at times it simply would not operate. This necessitated changing the voltage of the starter and generator to 12 volts, which has been standard practice, even though it meant an increase in battery weight of 10 lbs.

It might be of interest to mention that the problems encountered in starting the engines were the same in the 4 and 6 cylinder units—that is, while the 6 volt system would work satisfactorily in the laboratory, in the airplane on either the 4 or 6 cylinder engine, the hot starting was the major problem.

The subject of engine cooling brings up many interesting problems which have been encountered, and which apparently will face us in the future. The engine maker has left the problem of cooling the engine in general up to the airplane maker, who in many cases by training has not had the actual experience necessary to cool one of these modern engines. For that reason cooling problems enter in which require a great deal of revision at times by the engine maker in order to get the engines by the C.A.A. tests. It is felt that there should be somewhere in the aircraft industry a definite research program undertaken on the cooling of in-line or opposed engines, and from it obtain certain data which the various plane makers could apply to their designs. In many cases we have found that on a test stand an engine might average 400 degrees cylinder head temperature with 100 degrees air temperature under continuous full throttle power at the rated speed, where in the airplane the same engine might have cylinder temperatures of 550 degrees and only by most tricky baffling could the temperatures be taken back to anywhere near a safe temperature. It may be that we makers of light airplane engines will be forced to use blowers for cooling, in order to permit the airplane maker to submerge the engine in the fuselage or the wing and thus improve the aerodynamic characteristics of the airplane. If it is desired that the blower cooling be used, the cooling can be done by a power expenditure of approximately 4 per cent, providing there is sufficient space to permit the use of a blower of efficient proportions. In general, a blower of the backward curve type (see Fig. 5) will have an air efficiency of 50 per cent and can be produced very economically and reasonably light in weight—a matter of 3 lbs. and housings, for a 100 hp. engine. The blower, of course, must be very fragile to be sufficiently light and to operate continually without disintgration, must be driven through a rubber flexible member, preferably through rubber bushings. Our experience has been that a blower mounted directly on the crankshaft would

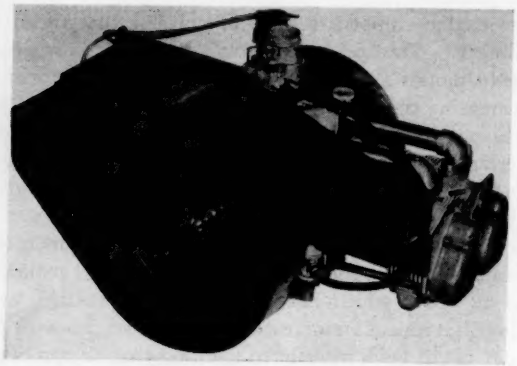

FIG. 5. Four-cylinder air-cooled engine with integral cooler, showing vacuum type of cooling fins.

fail after 10 to 15 hrs. of full throttle operation, whereas with a rubber drive it will last indefinitely.

In a previous paper on "Economic Aspects of Light Airplane Engines," the author called attention to the many parts which the Aircooled Motors Corporation was using having automotive standards; now approximately a year later, it is found that materials which would operate satisfactorily up to 137 lbs. brake mean effective pressure, will not operate satisfactorily in these engines which are hovering around 150 lbs. brake mean effective pressure. This is an indication that the automotive industry is lagging behind in the development of power plants; in fact, it would appear that the automotive engine makers could now draw on the light airplane engine makers for changes in design which would improve greatly the power output. The reason for this situation, no doubt, is due to the fact that the tooling of the automotive engine is nowhere near as flexible as the tooling of the light engine manufacturer. In other words, if the light engine manufacturer develops something noteworthy which would improve the product, it is a rather simple task to get it in production, whereas in the automotive industry a change must be considered from the immense amount of tooling which might be scrapped. A particular example of this situation is the fact that the present Ford V-8 engine was introduced in 1932 and the present Plymouth engine in 1933, the intervening years having been spent on development of these basic power plants and not in basic changes in designs involving scrapping of a vast tool program.

As the volume of the light airplane engine increases, and engine prices go lower and lower, the problem of tool obsolesence is going to be one which is more important and which may tend to slow up the frequent model changes of the light aircraft engine. It was with this thought in mind that the broad engine program was inaugurated in 1938 by the Aircooled Motors Corporation, feeling that even though power demands increased, the customers could be satisfied by one of the engines in the line without scrapping the tooling or the preceding engines.

In conclusion, it might be well also to discuss certain service problems which have been encountered. These small engines are required to accomplish a much more difficult task than any automobile engine, regardless of price, with very little definite servicing. For example: the author recommends that light airplane engines have a major overhaul at 500 hrs., or, in the small training ships, approximately 33,000 miles and in the faster ships 50,000 miles. To operate satisfactorily in an airplane these engines must maintain their power consistently during the entire 500 hrs. and not drop off more than 10 per cent; otherwise it would not be safe to fly the airplane. In an automobile the engine operates on an average of approximately 15 per cent maximum power and usually has a semi-overhaul of valves and rings at 25,000 miles. Without this overhaul the general output is usually found approximately 70 per cent maximum. In other words, even though the engine has fallen off in output of approximately 30 per cent, it still can drive the car safely or until the owner decides to have the engine overhauled.

Checking back on the average fuel consumption figures over 500 hrs. of operation, it was found that the average light airplane engine is developing 50 to 75 per cent power, thus subjecting the engine to much greater loads than the automobile engine. Furthermore, the automobile engine is sold by a nationwide sales organization which is consistently impressing on the owners the importance of service. In the light airplane field, servicing of the engine apparently is undertaken after there are indications that the engine is not operating normally. Instead of anticipating the service troubles, the troubles are actually present. This condition must be corrected before the light airplane can sell itself to the operators of automobiles as a safe proposition.

It is felt that the combined efforts of the airplane engine maker and the airplane maker will eventually result in making the airplane much more fool-proof to operate, even though it might necessitate a slight decrease in the power output; especially as regards the use of automatic heat control on the carbureter, which would eliminate the possibility of the engine failing to respond to the throttle after a long glide, and would also eliminate the possibility of ice formation on the carbureter and the subsequent possibility of engine failure. There are also indications that magneto ignition might be superseded with a battery type similar to that used on automobiles with somewhat better electrical qualities, by incorporating a spark advance curve which adapts itself to the particular power curve throughout the speed range and not to a particular speed, as is the speed range and not to a particular speed, as is the present practice with fixed spark magnetos. This would also permit the use of a vacuum part throttle mechanism which would increase the spark advance for cruising and in turn increase the economy.

A Simplified Method for Predicting the Change in Airplane Performance Due to a Change in Parameter

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SUMMARY

A method is presented for rapidly determining the effect of small modifications in the various parameters influencing performance. The only requirements for use of this method are performance values for some basic condition and the corresponding airplane polar curve. No recourse is necessary to analytical methods of performance calculation, but only the fundamental concepts of power available and power required are used.

The resulting expressions lead to some interesting corollaries dealing with critical values of the parameters.

INTRODUCTION

THERE HAS been a need for some method of rapidly estimating the effect of minor changes in airplane characteristics, such as weight, wing area, power, etc., on the flight performance. The various analytical methods of performance prediction generally include means for determining the effect of change in parameters but these are of little value to the aerodynamist who has already determined the performance by means of conventional power required and power available curves. For example, equations are shown[1, 2] for variations in performance but these are in terms of certain parameters which are in turn functions of effective span, parasite and power loading. The development of the equations in these references is contingent on the determination of a parabolic equation to represent the polar curve. While this could be done with good accuracy in most cases, it would require additional work here and the use of assumptions not consistent with the original analysis. The only recourse for the designer is to recalculate the power curves and determine new values independently of the original. Obviously, this procedure also has its shortcomings inasmuch as the errors inherent in the graphical method may be greater than the small differences in performance involved, not to mention the labor of recalculating and replotting the curves.

In this paper, a method is developed for obtaining the change in performance without recourse to any special system of performance analysis. It is only necessary to have a plot of the polar curve on which the original performance calculations are based. The variation of this curve from a parabolic does not affect the method.

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BASIC CONSIDERATIONS

All items of performance can be referred back to the plot of lift *versus* drag coefficient for the airplane, this being known as the airplane polar curve. It is necessary to have such a curve available before the basic calculations can be made, regardless of whether it is taken directly from a wind tunnel test or built up from estimated data. To obtain expressions for the rate of change of performance with any parameter, it is then only necessary to determine the effect of this change on the polar curve and the corresponding effect of the polar curve modification on the airplane performance. Following this line of procedure for various items of performance, the resulting expressions all come out in terms of the function $(dC_D/dC_L)(C_L/C_D)$ which depends only on the shape of the polar curve. This function, henceforth called β, when once plotted against indicated velocity for the basic design, can be used for all subsequent minor modifications to the design. Fig. 1

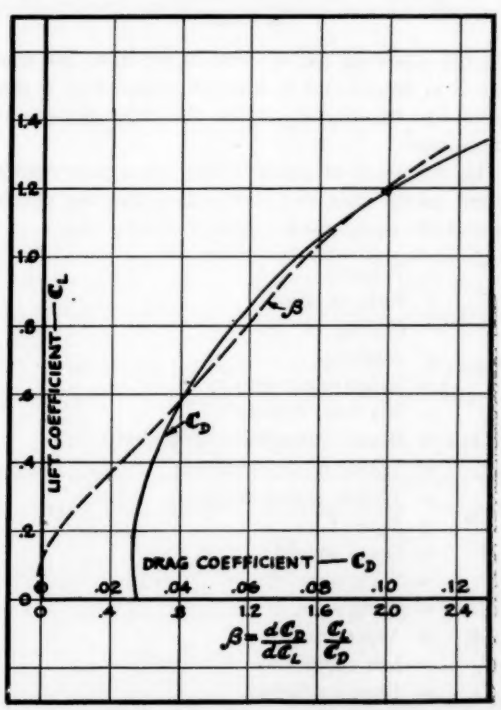

FIG. 1. Typical polar showing variation of β with C_L.

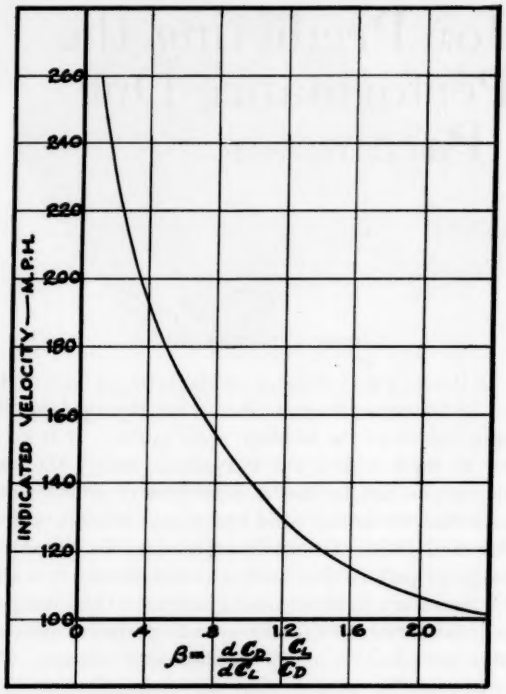

FIG. 2. Typical variation of β with indicated velocity.

shows a typical plot of β *versus* lift coefficient, C_L, while Fig. 2 shows this plotted against indicated velocity, V_i.

SYMBOLS

In the following list of symbols, no units are shown since it is immaterial as long as consistency is maintained. In the final equations, the ratios become nondimensional.

The definition of parasite drag used here does not include profile drag in order to facilitate the development of the equations for change in wing area.

V = Velocity
C = Rate of climb
H = Ceiling
h = Altitude
σ = Relative air density
ρ_0 = Sea level density
b.hp. = Brake horsepower available
P_a = Thrust power available
P_r = Thrust power required
$\Delta P = P_a - P_r$
W = Gross weight
S = Wing area
b = Wing span
AR = Aspect ratio
C_L = Lift coefficient
C_D = Drag coefficient
C_{D_0} = Profile drag coefficient
C_{D_p} = Parasite drag coefficient (exclusive of wing drag)
C_{D_i} = Induced drag coefficient
$C_{D_w} = C_{D_0} + C_{D_i}$ = Wing drag coefficient
dC_D' = Increment in C_D at constant C_L
dC_D = Increment in C_D due to increment in C_L
$\beta = (dC_D/dC_L)(C_L/C_D)$
k = Miscellaneous constant

SAMPLE DERIVATION

To illustrate the derivation of the various formulas for performance variation, a typical example is worked out here, namely, the effect of change in power available on level flight speed. This derivation is presented to demonstrate the general method of attacking the problem. The same procedure may be used for determining expressions for other items of performance and other parameters. The equation for speed is

$$V = [P_a/(C_D S\sigma\rho_0/2)]^{1/3}$$

Taking the logarithmic derivative of both sides of this equation,

$$dV/V = (1/3)(dP_a/P_a)$$

This indicates that the velocity varies one-third as fast as the power, when expressed in per cent of the original values. This may be recognized as the familiar cubic speed-power relationship. However, in taking the logarithmic derivative, the drag coefficient, C_D, was assumed constant. At high speeds, this assumption is justified but at lower speeds, the drag coefficient may vary considerably with small changes in velocity. It is therefore necessary to consider C_D as a variable also and the logarithmic derivative now becomes

$$dV/V = (1/3)(dP_a/P_a - (1/3)(dC_D/C_D)$$

C_D depends only on C_L which in turn depends on V. The above term involving C_D can therefore be evaluated as follows:

$$\frac{dC_D}{C_D} = \frac{dC_D}{dC_L}\frac{dC_L}{dV}\frac{dV}{C_D}$$

But

$$C_L = W/(SV^2\sigma\rho_0/2) \text{ and } dC_L/dV = -2(C_L/V)$$

Hence

$$\frac{dC_D}{C_D} = -2\frac{dC_D}{dC_L}\frac{C_L}{V}\frac{dV}{C_D} = -2\beta\frac{dV}{V}$$

where

$$\beta = (dC_D/dC_L)(C_L/C_D)$$

Substituting

$$dV/V = (1/3)(dP_a/P_a) + (2/3)\beta(dV/V)$$
$$= \{(1/3)/[1 - (2/3)\beta]\}(dP_a/P_a) \qquad \text{(A)}$$

A new factor is now included, namely, $1 - (2/3)\beta$, which takes into account the variation of C_D with V. To illustrate the importance of this factor, two conditions are assumed: one at high speed where the slope of the polar curve is nearly vertical and the other at the point of best L/D where the tangent to the polar curve passes through the origin. In the first case, β is approximately zero and Eq. (A) becomes

$$dV/V = (1/3)(dP_a/P_a)$$

For the second case, $\beta = 1$ and Eq. (A) becomes

$$dV/V = dP_a/P_a$$

It is apparent from this illustration that the cubic speed-power relation would give very erroneous results if applied near best L/D.

If the polar curve can be represented by a parabolic equation, then the term β becomes equal to twice the parameter Γ_σ of reference 2. The formulas of part E of that reference are then equivalent to those derived here. This method can be looked on as a generalized version of reference 2, applying to any polar curve while the latter applies only to a parabolic. β can also be considered as a graphical interpretation of $2\Gamma_\sigma$.

Accuracy

Eq. (A) is rigorously correct only for infinitesimal variations in the parameter since differentials are used. However, for practical purposes, it may be applied for finite changes up to considerable percentages, depending on the form of the β function in the region considered. In the high-speed range, β is nearly constant and, therefore, Eq. (A) holds for large variations. At lower speeds, β varies considerably and, therefore, the formula cannot be used here for too large a change in parameter. However, β enters into the expression as a secondary factor and this tends to minimize its effect on the accuracy of the results.

The propeller efficiency is assumed constant for the increments in performance dealt with. This assumption is reasonable for small variations in parameter, but may introduce some error if too large a change is taken. If the rate of change of propeller efficiency with parameter is known, this can be included in the expressions for performance change but, in general, the added complication is not justified.

Generalized Formula

The same procedure as used in the development of Eq. (A) can be used to determine other expressions for rate of change of performance with various parameters. However, before undertaking the derivation of further formulas, it is convenient to generalize on the mathematics involved.

Consider the function

$$z = cx^m y^n$$

Taking logarithms

$$\log z = \log c + m \log x + n \log y$$

Taking differentials

$$dz/z = m(dx/x) + n(dy/y)$$

Now consider the case where

$$y = f(u) \text{ and } u = kx^a z^b$$

Hence

$$\frac{dz}{z} = m\frac{dx}{x} + n\frac{dy}{du}\left(\frac{\partial u}{\partial x}\frac{dx}{y} + \frac{\partial u}{\partial z}\frac{dz}{y}\right)$$

$$= m\frac{dx}{x} + n\frac{dy}{du}\left(a\frac{u}{x}\frac{dx}{y} + b\frac{u}{z}\frac{dz}{y}\right)$$

$$= \frac{m + na(dy/du)(u/y)}{1 - nb(dy/du)(u/y)}\frac{dx}{x}$$

If $y = f(u)$ could be expressed analytically and z be solved as an explicit function of x, the same result could be arrived at directly without recourse to partial differentiation. However, in the cases to be dealt with, this cannot be done and it is necessary to use the method outlined above.

Density-Altitude and Power-Altitude Relations

Before proceeding with the development of general formulas, it is first necessary to determine an analytical relation between density and altitude, and power and altitude.

From standard atmosphere data, the density ratio is given in terms of standard altitude as follows:

$$\sigma = (1 - 6.88 \times 10^{-6}h)^{4.256}$$

Hence

$$\frac{d\sigma}{\sigma} = \frac{-29.3 \times 10^{-6}h}{1 - 6.88 \times 10^{-6}h}\frac{dh}{h}$$

$$= -\Psi(h)(dh/h)$$

The function $\Psi(h)$ is calculated for different altitudes and is plotted in Fig. 3. This relation applies only up to the lower level of the isothermal atmosphere, *i.e.*, 35,332 feet.

The variation of brake horsepower with altitude at constant r.p.m. can be approximated by the relation

$$\text{b.hp.} = \text{b.hp.}_0\, \sigma^{1.23}$$

where the density ratio, σ, is taken to correspond with standard altitude. Taking the logarithmic derivative and using the above relation between σ and h

$$d\text{ b.hp./b.hp.} = dP_a/P_a = 1.23(d\sigma/\sigma) = -1.23\Psi(h)(dh/h)$$

Level Flight Speed

In this section, the development of formulas for change of velocity due to change in various parameters

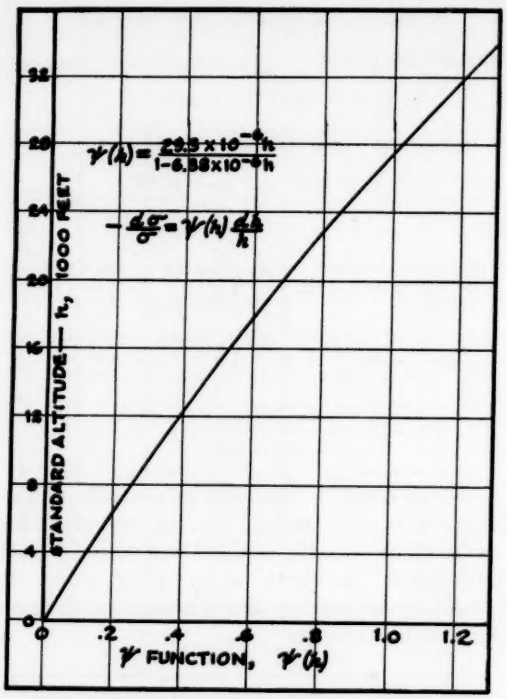

FIG. 3. Ψ function of standard altitude.

is shown. For each parameter change, all others are considered constant, that is, only one parameter at a time is considered a variable. An exception to this is the variation of wing area at constant aspect ratio when both area and span are considered to vary simultaneously. Likewise, the variation of altitude above the critical includes the corresponding variation of engine power. The change in parameter σ is assumed independent of power and is included mainly for purposes of flight test reduction.

The relations between speed and the various parameters are first given in their implicit forms and these relations are used in the subsequent partial differentiations:

$$V = [P_a/(C_D S\sigma\rho_0/2)]^{1/3}$$

$$C_L = W/(SV^2\sigma\rho_0/2)$$

$$C_D = C_{D_0} + C_{D_p} + C_{D_i}$$

$$C_{D_p} = k/S$$

$$C_{D_i} = C_L^2 S/\pi b^2$$

Change in weight = dW

$$\frac{dV}{V} = -\frac{1}{3}\frac{dC_D}{dC_L}\left(\frac{\partial C_L}{\partial W}\frac{dW}{C_D} + \frac{\partial C_L}{\partial V}\frac{dV}{C_D}\right) = -\frac{1}{3}\frac{dC_D}{dC_L}\left(\frac{C_L}{W}\frac{dW}{C_D} - 2\frac{C_L}{V}\frac{dV}{C_D}\right)$$

$$= \frac{-(1/3)\beta}{1-(2/3)\beta}\frac{dW}{W} \qquad (1)$$

Change in drag coefficient = dC_D'

$$\frac{dV}{V} = -\frac{1}{3}\frac{dC_D'}{C_D} - \frac{1}{3}\frac{dC_D}{dC_L}\frac{dC_L}{dV}\frac{dV}{C_D}$$

$$= \frac{-1/3}{1-(2/3)\beta}\frac{dC_D'}{C_D} \qquad (2)$$

Change in power = dP_a

$$\frac{dV}{V} = \frac{1}{3}\frac{dP_a}{P_a} + \frac{2}{3}\frac{dC_D}{dC_L}\frac{C_L}{V}\frac{dV}{C_D}$$

$$= \frac{1/3}{1-(2/3)\beta}\frac{dP_a}{P_a} \qquad (3)$$

Change in wing area = dS

$$\frac{dV}{V} = -\frac{1}{3}\frac{dS}{S} - \frac{1}{3}\left[\frac{\partial C_D}{\partial C_L}\left(\frac{\partial C_L}{\partial S}\frac{dS}{C_D} + \frac{\partial C_L}{\partial V}\frac{dV}{C_D}\right) + \frac{\partial C_D}{\partial S}\frac{dS}{C_D}\right]$$

$$= \frac{-(1/3)[1-\beta+(C_{D_i}{}^* - C_{D_p})/C_D]}{1-(2/3)\beta}\frac{dS}{S} \qquad (4)$$

Change in span = db

$$\frac{dV}{V} = -\frac{1}{3}\left(\frac{\partial C_D}{\partial b}\frac{db}{C_D} + \frac{\partial C_D}{\partial C_L}\frac{dC_L}{dV}\frac{dV}{C_D}\right)$$

$$= \frac{2/3}{1-(2/3)\beta}\frac{C_{D_i}}{C_D}\frac{db}{b} \qquad (5)$$

If the span and area change in such a manner that the aspect ratio remains constant, Eq. (5) is altered as follows:

$$b = (\pi\,\mathrm{AR}\,S)^{1/2}$$

$$db/b = (1/2)(dS/S)$$

$$\frac{dV}{V} = \frac{1/3}{1-(2/3)\beta}\frac{C_{D_i}}{C_D}\frac{dS}{S}$$

Change in relative air density = $d\sigma$

$$\frac{dV}{V} = -\frac{1}{3}\frac{d\sigma}{\sigma} - \frac{1}{3}\frac{dC_D}{dC_L}\left(\frac{\partial C_L}{\partial \sigma}\frac{d\sigma}{C_D} + \frac{\partial C_L}{\partial V}\frac{dV}{C_D}\right)$$

$$= \frac{-(1/3)(1-\beta)}{1-(2/3)\beta}\frac{d\sigma}{\sigma} \qquad (6)$$

Change in standard altitude = dh

$$d\sigma/\sigma = -\Psi(h)(dh/h)$$

(a) If below critical altitude

P_a = constant

$$\frac{dV}{V} = \frac{(1/3)(1-\beta)}{1-(2/3)\beta}\Psi(h)\frac{dh}{h} \qquad (7)$$

(b) If above critical altitude

* If the aspect ratio is constant, the corresponding change in span cancels out the induced drag term (Eq. (4')).

$$dP_a/P_a = 1.23(d\sigma/\sigma)$$

$$\frac{dV}{V} = \frac{-(1/3)(1-\beta)}{1-(2/3)\beta}\frac{d\sigma}{\sigma} + \frac{1/3}{1-(2/3)\beta}\frac{dP_a}{P_a}$$

$$= \frac{-(1/3)(0.23+\beta)}{1-(2/3)\beta}\Psi(h)\frac{dh}{h} \qquad (7')$$

Rate of Climb

Variations in rate of climb are all taken at constant air speed. At best rate of climb, the rate of change of power available and power required with velocity are equal and, therefore, small changes in climbing speed have no effect on best rate of climb.

$$C = (P_a - P_r)/W = \Delta P/W$$

$$P_r = C_D S V^3 \sigma \rho_0/2$$

Change in weight = dW

$$\frac{dC}{C} = -\frac{dW}{W} - \frac{dP_r}{dC_D}\frac{dC_D}{dC_L}\frac{dC_L}{dW}\frac{dW}{\Delta P}$$

$$= -\frac{dW}{W} - \frac{P_r}{C_D}\frac{dC_D}{dC_L}\frac{C_L}{W}\frac{dW}{\Delta P}$$

$$= -(1+\beta P_r/\Delta P)(dW/W) \qquad (8)$$

Change in drag coefficient = dC_D'

$$\frac{dC}{C} = -\frac{dP_r}{dC_D}\frac{dC_D'}{\Delta P} = -\frac{P_r}{\Delta P}\frac{dC_D'}{C_D} \qquad (9)$$

Change in power = dP_a

$$\frac{dC}{C} = \frac{dP_a}{\Delta P} = \frac{P_a}{\Delta P}\frac{dP_a}{P_a} \qquad (10)$$

Change in wing area = dS

$$\frac{dC}{C} = \frac{\partial P_r}{\partial S}\frac{dS}{\Delta P} - \frac{\partial P_r}{\partial C_D}\left(\frac{\partial C_D}{\partial C_L}\frac{dC_L}{dS}\frac{dS}{\Delta P} + \frac{\partial C_D}{\partial S}\frac{dS}{\Delta P}\right)$$

$$= -\frac{P_r}{\Delta P}\left(1-\beta+\frac{C_{D_i}{}^* - C_{D_p}}{C_D}\right)\frac{dS}{S} \qquad (11)$$

Change in span = db

$$\frac{dC}{C} = -\frac{dP_r}{dC_D}\frac{dC_D}{db}\frac{db}{\Delta P} = 2\frac{P_r}{C_D}\frac{C_{D_i}}{b}\frac{db}{\Delta P}$$

$$= 2\frac{P_r}{\Delta P}\frac{C_{D_i}}{C_D}\frac{db}{b} \qquad (12)$$

If the span and area change in such a manner that the aspect ratio remains constant, Eq. (12) is altered as follows:

$$dC/C = (P_r/\Delta P)(C_{D_i}/C_D)(dS/S)$$

Change in relative air density = $d\sigma$

$$\frac{dC}{C} = -\frac{\partial P_r}{\partial \sigma}\frac{d\sigma}{\Delta P} - \frac{\partial P_r}{\partial C_D}\frac{dC_D}{dC_L}\frac{dC_L}{d\sigma}\frac{d\sigma}{\Delta P}$$

$$= -\frac{P_r}{\Delta P}(1-\beta)\frac{d\sigma}{\sigma} \qquad (13)$$

Change in standard altitude = dh

$$d\sigma/\sigma = -\Psi(h)(dh/h)$$

(a) If below critical altitude

P_a = constant

$$dC/C = (P_r/\Delta P)(1-\beta)\Psi(h)(dh/h) \qquad (14)$$

(b) If above critical altitude

$$dP_a/P_a = 1.23(d\sigma/\sigma)$$

$$\frac{dC}{C} = \frac{-1.23P_a - (1-\beta)P_r}{\Delta P}\Psi(h)\frac{dh}{h} \qquad (14')$$

Ceiling

The criterion for absolute ceiling is the equality of power available and power required. Hence, the differential equation for change of ceiling can be set up by equating the change in power available with the change in power required. The ceiling is assumed above the critical altitude of the engines and therefore, the power-altitude relation previously developed is used. Ceiling occurs at maximum rate of climb and, hence, change in climbing speed has no effect.

$$P_r = C_D S V^3 \sigma \rho_0/2$$

$$\text{b.hp.} = \text{b.hp.}_0\sigma^{1.23}$$

$$(dP_a/P_a) = (dP_r/P_r)$$

Change in weight = dW

$$1.23\frac{d\sigma}{\sigma} = \frac{d\sigma}{\sigma} + \frac{dC_D}{dC_L}\left(\frac{\partial C_L}{\partial W}\frac{dW}{C_D} + \frac{\partial C_L}{\partial \sigma}\frac{d\sigma}{C_D}\right) =$$

$$\frac{d\sigma}{\sigma} + \beta\frac{dW}{W} - \beta\frac{d\sigma}{\sigma}$$

$$\Psi(H)\frac{dH}{H} = \frac{-\beta}{0.23+\beta}\frac{dW}{W} \qquad (15)$$

Change in drag coefficient = dC_D'

$$1.23\frac{d\sigma}{\sigma} = \frac{d\sigma}{\sigma} + \frac{dC_D'}{C_D} - \frac{dC_D}{dC_L}\frac{C_L}{\sigma}\frac{d\sigma}{C_D}$$

$$\Psi(H)\frac{dH}{H} = \frac{-1}{0.23+\beta}\frac{dC_D'}{C_D} \qquad (16)$$

Change in power = dP_a

$$\frac{dP_a}{P_a} + 1.23\frac{d\sigma}{\sigma} = \frac{d\sigma}{\sigma} - \frac{dC_D}{dC_L}\frac{C_L}{\sigma}\frac{d\sigma}{C_D}$$

$$\Psi(H)(dH/H) = [1/(0.23+\beta)](dP_a/P_a) \qquad (17)$$

Change in wing area = dS

$$1.23\frac{d\sigma}{\sigma} = \frac{d\sigma}{\sigma} + \frac{dS}{S} + \frac{\partial C_D}{\partial C_L}\left(\frac{\partial C_L}{\partial S}\frac{dS}{C_D} + \frac{\partial C_L}{\partial \sigma}\frac{d\sigma}{C_D}\right) +$$

$$\frac{\partial C_D}{\partial S}\frac{dS}{C_D}$$

* If the aspect ratio is constant, the corresponding change in span cancels out the induced drag term (Eq. 11′)).

$$\Psi(H)\frac{dH}{H} = -\frac{[1-\beta+(C_{D_i}{}^{*}-C_{D_p})/C_D]}{0.23+\beta}\frac{dS}{S} \quad (18)$$

Change in span = db

$$1.23\frac{d\sigma}{\sigma} = \frac{d\sigma}{\sigma} + \frac{\partial C_D}{\partial b}\frac{db}{C_D} + \frac{\partial C_D}{\partial C_L}\frac{dC_L}{d\sigma}\frac{d\sigma}{C_D}$$

$$\Psi(H)\frac{dH}{H} = \frac{2}{0.23+\beta}\frac{C_{D_i}}{C_D}\frac{db}{b} \quad (19)$$

If the span and area change in such a manner that the aspect ratio remains constant, Eq. (19) is altered as follows:

$$\Psi(H)\frac{dH}{H} = \frac{1}{0.23+\beta}\frac{C_{D_i}}{C_D}\frac{dS}{S}$$

Performance Involving Integrated Values

Performance items, such as time to climb, range, endurance, takeoff and landing are integrated functions and, therefore, it is not possible to obtain simple expressions for change in performance. However, reasonable accuracy may be obtained by using average values of the variables involved and assuming that the functions vary in proportion to these.

As an example, time to climb is the integral of the increment in altitude divided by the rate of climb. To solve for the change in time to climb, the change in the average rate of climb is determined. Likewise, range, which is the integral of weight increment times miles per unit fuel weight, is assumed equal to an average miles per unit fuel weight times the total change in weight.

Corollaries

As a check on the accuracy of the formulas and an illustration of their use in determining critical values of the parameters, the following corollaries are derived:

(*1*) *Minimum Power Required.* All the formulas for change in speed have the term $1-(2/3)\beta$ in the denominator. When this term vanishes, the rate of change of velocity with parameter is infinite, or inversely, the rate of change of parameter with velocity is zero. Expressing the latter as the derivative of power required with respect to velocity divided by the derivative of power required with respect to parameter, it is evident that the numerator must be zero if the denominator is not. It is shown in the ensuing corollaries that the singular cases where the rate of change of power required with parameter is zero occur at points on the polar curve considerably below the region considered here. Hence, it can be concluded that the rate of change of power with velocity is zero here, thus defining the minimum power required point. Solving for the corresponding value of β,

$$1-(2/3)\beta = 0 \quad \beta = 3/2 = (dC_D/dC_L)(C_L/C_D)$$

$$dC_D/dC_L = (3/2)(C_D/C_L)$$

The lift coefficient corresponding to this relation is located graphically on the polar curve of Fig. 4.

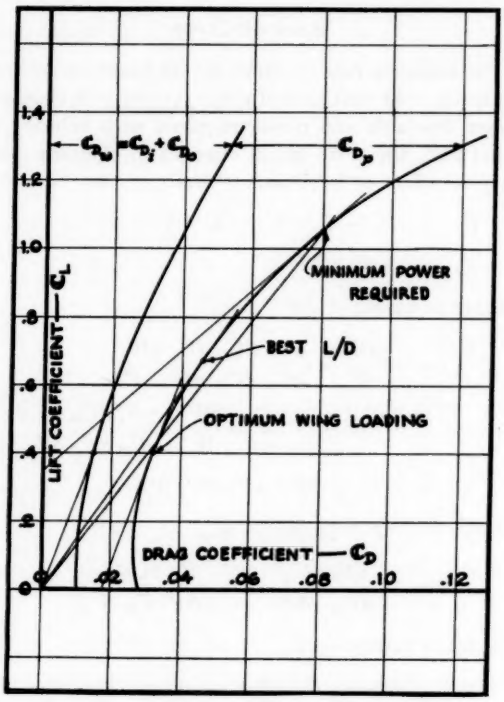

FIG. 4. Typical wing and airplane polar showing critical points.

(*2*) *Optimum Wing Loading.* The optimum wing loading is obtained by varying the wing area until minimum total drag is obtained, assuming constant aspect ratio. At this point, for a given power available, change in wing area results in no change in speed, rate of climb, or ceiling. Hence, equating either of Eqs. (4′), (11′) or (18′) to zero,

$$1-\beta-(C_{D_p}/C_D) = 0$$

$$\beta = (C_D - C_{D_p})/C_D = C_{D_w}/C_D$$

$$(dC_D/dC_L)(C_L/C_D) = (C_{D_w}/C_D)$$

$$dC_D/dC_L = C_{D_w}/C_L$$

The point on the polar curve satisfying this relation determines the optimum wing loading. At lift coefficients below this value, a decrease in wing area results in increased performance while above it the reverse effect occurs. This accounts for the fact that an increase in wing loading for a given gross weight usually results in an increase in high speed, but a decrease in best rate of climb, the lift coefficients for these two conditions being on opposite sides of the optimum value.

* If the aspect ratio is constant, the corresponding change in span cancels out the induced drag term (Eq. (18′)).

The point corresponding to the condition $dC_D/dC_L = C_{D_w}/C_D$ can be determined graphically by locating a lift coefficient where the slope of the airplane polar curve equals the L/D of the wing-alone polar curve. This is shown in Fig. 4. It can be seen from this construction that the optimum wing loading point becomes coincident with the wing-alone best L/D if the parasite drag coefficient, C_{D_p} is constant.

(3) *Altitude for Minimum Drag.* In a similar manner to the determination of optimum wing loading, the formulas for rate of change of speed and rate of climb with altitude are equated to zero to determine the optimum altitude. The resulting condition is

$$1 - \beta = 0$$

$$dC_D/dC_L = C_L/C_D$$

This corresponds to the best L/D point on the polar curve and, therefore, to minimum drag for a given weight. Below the lift coefficient corresponding to this point, an increase in altitude at constant power available results in an increase in level flight speed or rate of climb, while above it, the reverse takes place. This critical point then determines the intersection of power required curves for adjacent altitudes. It also locates the altitude beyond which supercharging is ineffective, that is, beyond which there is no increase in speed at constant power.

REFERENCES

[1] Oswald, W. B., *General Formulas and Charts for the Calculation of Airplane Performance*, N.A.C.A. Technical Report No. 408, 1932.

[2] Rockefeller, W. C., *General Airplane Performance*, N.A.C.A. Technical Report No. 654, 1939.

Letter to the Editor

July 2, 1941

Dear Sir:

In my letter to you of March 14, 1941, which was published in the April 1941, Journal, I pointed out that the frequency ratio ω_β/ω_h for movable control surfaces is mostly smaller than one excepting symmetrical wing aileron oscillations. Also Jean Wylie[3] uses $\omega_\beta/\omega_h = 0$ for almost all his charts. I further pointed out that W. B. Bergen and Lee Arnold,[1] drew all curves in their four diagrams of Fig. 19 with asymptotes $\omega_\beta/\omega_h = 1$ as minimum value.

I did not state that values $\omega_\beta/\omega_h < 1$ for symmetrical wing bending-aileron were mostly encountered, nor did I originate the idea of asymptotes $\omega_\beta/\omega_h = 1$ shown in Fig. 19,[1] as given in the reply of the authors published in the June 1941, issue of the Journal. If Bergen and Arnold now state that "the curves exhibit a sharp change in slope at $\omega_\beta/\omega_h = 1$ and cross the $\omega_\beta/\omega_h = 0$ axis at a finite value of x_β instead of approaching an asymptote at $\omega_\beta/\omega_h = 1$," they should correct these in their Fig. 19. The authors further point out that in their numerical example only a value of $r_\beta^2 = 1/1000$ was used, while Theodorsen's[2] graph II-B has $r_\beta^2 = 1/120$, and that flutter would occur for the smaller value r_β^2. The influence of r_β^2 can best be seen from Theodorsen's[2] graph II-A, d on page 17, which shows a reduction in flutter range for smaller values r_β^2. Therefore, the conclusion that no flutter occurs appears quite justified from this graph. However, there is the possibility that not a sufficient number of points were calculated to prove that the loops of the Theodorsen curves do not extend into the $\omega_\beta/\omega_h > 1.2$ region, or that a reverse in trend might occur at very small values r_β^2. To clarify this a set of calculations was run in the manner explained by Theodorsen[2] using exactly the same parameters as in the authors' "numerical example" on page 5.[1] The following values for "Reals and Imaginaries" were obtained as plotted in Fig. 1.

	$1/k=$	0	0.25	0.50	1	1.51516	2.00	3.333
$\sqrt{\bar{X}}$	Reals	3.92	3.85	3.66	3.18	2.926	2.81	2.62
		2.02	1.94	1.60				
	Im.	3.106	3.106	3.098	3.064	2.979	2.829	1.698

It can be seen that this is a borderline case since the curves barely intersect. The ever present frictional and structural damping will prevent flutter in this case. The advantage of presenting

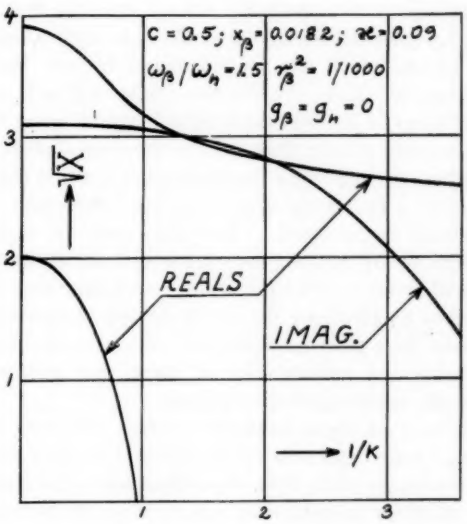

FIG. 1.

the results in the Theodorsen[2]-Wylie[3] manner becomes evident: Not only is the flutter speed parameter directly obtained from the physical properties of the oscillating structure for the benefit of the designer, but also from the intersection with the loops of their curves it can easily be seen whether flutter is only a borderline case or of a more violent nature.

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[1] Bergen, W. B., and Arnold, Lee, *Graphical Solution of the Bending-Aileron Case of Flutter*, Journal of the Aeronautical Sciences, Vol. 7, No. 12, page 495, October, 1940.

[2] Theodorsen, Th., and Garrick, I. E., *Mechanism of Flutter*, N.A.C.A. Technical Report No. 685, 1940.

[3] Wylie, Jean, *Perpendicular Axes Control Surface Binary Flutter*, C.A.A. Report No. 23 April, 1941.

The Effect of Wind Tunnel Wall Interference on the Stalling Characteristics of Wings

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ABSTRACT

In this paper it is shown that the distortion of the lift distribution of a wing in a wind tunnel caused by the wind tunnel wall interference can be considered as an induced twist in the wing. The magnitude of this twist is calculated for a wing in a circular closed wind tunnel. In numerical examples and an experimental check it is seen that this induced twist (washin for a closed wind tunnel working section) is large enough to cause very marked changes in the stalling characteristics of a wing unless the wing span is less than eighty per cent of the wind tunnel diameter.

THE ALMOST universal use of highly tapered and highly loaded monoplane wings has made the problem of stalling characteristics one of great importance. A wing with a tendency toward tip-stall will almost invariably have undesirable lateral stability and control characteristics near the stall. The normal induction effect, which can be calculated by the Prandtl wing theory, causes the effective angle of attack of the tip sections of an untwisted highly tapered wing to be considerably greater than that of the central portion of the wing; consequently the stalling angle is first reached at the wing tips as the angle of attack is increased; and a tip-stall is produced. This effect may be counteracted either by twisting the wing so that the geometrical angle of attack of the tip sections is less than that of the center or by changing the airfoil section so that the tip sections have more camber than the central portion of the wing. A combination of these two methods is normally used to prevent tip-stall.

As both of these methods increase the wing drag, the optimum design uses only sufficient twist or camber to prevent tip-stall. The theoretical methods of solving this problem are generally satisfactory; however, it is advisable to obtain an experimental check of the results in a model test. Wind tunnel investigations of the stalling of tapered wings, utilizing both force measurements and floss tufts to show the flow over the wing, have been carried out and described by Nazir[1] and by C. B. Millikan,[2] and such tests are now generally included in any wind tunnel test program. In all such tests which the author has observed, the distortion of the lift distribution due to the interference of the wind tunnel walls has been neglected. Although this distortion is generally of little significance below the stall, it will be shown in the present paper to be of considerable importance in connection with the spanwise development of the stall. The pertinent calculation will be carried out for the case of a closed wind tunnel having a circular cross-section; similar results would be found for any other shape of closed wind tunnel working section.

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The general problem of the lift distribution for a wing of arbitrary plan form in a circular wind tunnel has been solved by C. B. Millikan.[3] His method is very similar to that developed by Lotz[4] for determining the lift on a wing in an unbounded fluid. For the present investigation it will, however, be sufficient to consider the much simpler inverse problem of determining the plan form necessary to produce a given lift distribution.

In the analysis, the following notations will be used:

Γ	=	Circulation about the wing (in the wind tunnel)
Γ_0	=	Circulation about the center of the wing (in the wind tunnel)
C_L	=	Lift coefficient (in the wind tunnel)
U	=	Velocity of airstream
w'	=	Downwash velocity due to wind tunnel wall interference
S	=	Wing area
b	=	Wing span
AR	=	b^2/S = Wing aspect ratio
r	=	Wind tunnel radius
x	=	Distance along the wing span from the center of the wing, positive to the right
α	=	Angle of attack corrected for wind tunnel wall interference
α_i	=	C_L/πAR = Self-induced angle of attack
α'_i	=	w'/U = Induced angle of attack due to wind tunnel wall interference
$\Delta\alpha'_i$	=	Induced washin of wing due to wind tunnel wall interference

If a wing in a wind tunnel has an elliptic lift distribution as in Fig. 1, the circulation is

$$\Gamma = \Gamma_0\sqrt{1 - (2x/b)^2}$$

The downwash at the wing due to the wind tunnel wall interference can be calculated from the image vortex system and is

$$w' = \frac{\Gamma_0}{\pi b}\int_{-b/2}^{b/2}\frac{\xi^2 d\xi}{(x\xi - r^2)\sqrt{b^2 - 4\xi^2}} = \frac{\Gamma_0 r^2}{2b}\left\{\frac{1}{x^2} - \frac{2r^2}{x^2\sqrt{4r^4 - b^2x^2}}\right\}$$

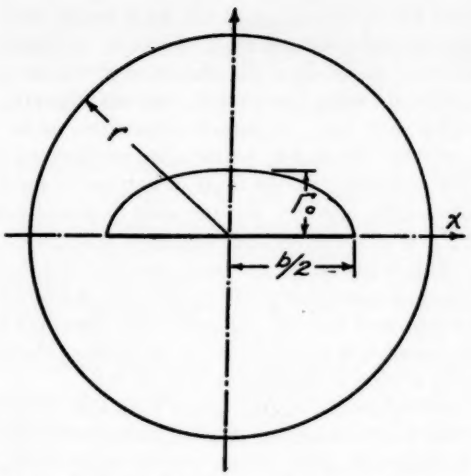


FIG. 1. Lift distribution for a wing in a wind tunnel.

The induced angle of attack due to the wind tunnel wall effect is thus

$$\alpha'_i = \frac{\Gamma_0 r^2}{2Ub}\left\{\frac{1}{x^2} - \frac{2r^2}{x^2\sqrt{4r^4 - b^2x^2}}\right\}$$

Since

$$\alpha_i = C_L/\pi\mathrm{AR} = \Gamma_0/2Ub$$

the induced downwash can be more conveniently written as

$$\frac{\alpha'_i}{\alpha_i} = \frac{r^2}{x^2}\left\{1 - \frac{2r^2}{\sqrt{4r^4 - b^2x^2}}\right\}$$

This downwash ratio is plotted in Fig. 2 as a function of position along the span of the wing for several values of the ratio of wing span to wind tunnel diameter.

From Fig. 2 it is apparent that the downwash due to wind tunnel wall effect may vary considerably over the span of a model wing. The magnitude of this variation is best seen from a numerical example. Consider a wing having the following characteristics:

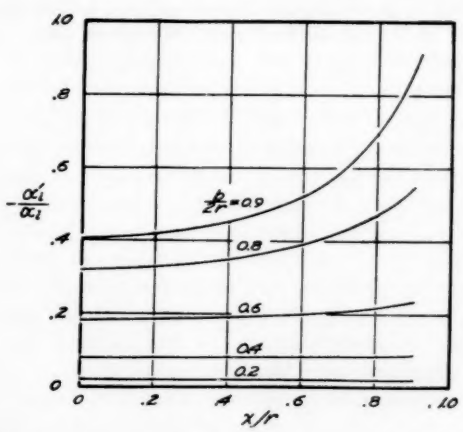


FIG. 2. Induced angle of attack due to wind tunnel wall interference.

$$b/2r = 0.9 \qquad C_L = 1.2 \qquad \mathrm{AR} = 7$$

Then $\alpha'_i = -2.71°$ at the wing tip, and $\alpha'_i = -1.27°$ at the center of the wing.

This means that such a wing of elliptic planform which has 1.44° of washout will behave when tested in a closed wind tunnel as though it had no twist when $C_L = 1.2$, while at higher lift coefficients the wing will behave as though it had washin. Such a wing would almost certainly have satisfactory stall characteristics in free air, but it would probably show tip-stall in the closed wind tunnel.

The effect of the wind tunnel wall interference on the lift distribution can thus be considered as an induced washin. The amount of washin induced by the tunnel walls, $\Delta\alpha'_i$, is the difference in α'_i at the center of the wing and at the wing tip. From the above expression for α'_i/α_i, the induced washin is

$$\frac{\Delta\alpha'_i}{\alpha_i} = \frac{4r^2}{b^2}\left\{\left[1 - \frac{b^4}{16r^4}\right]^{-1/2} - 1 - \frac{b^4}{32r^4}\right\}$$

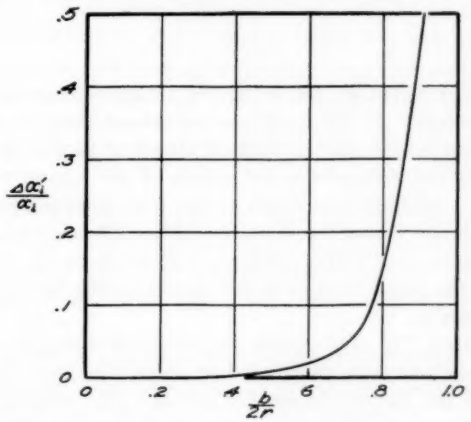


FIG. 3. Induced twist due to wind tunnel wall interference.

The induced washin is plotted in Fig. 3. This induced washin has been calculated for the case where the lift distribution is elliptical; however, any other reasonable lift distribution would give very nearly the same results. If a more accurate analysis is required for a particular wing, it can be carried out by using Millikan's method.

The results of Fig. 3 are qualitatively verified by wind tunnel test. A wing having an aspect ratio of 5.88 and a maximum lift coefficient of about 1.25 was tested in two different wind tunnels having $b/2r = 0.93$ and 0.77. The lift of the wing in the two different wind tunnels is compared in Fig. 4. In the larger wind tunnel, the wing equipped with tufts showed no signs of stall below an angle of attack of 17.3°.* At this angle the area near the inboard end of the aileron shows the beginning of the stall. When the angle of attack was

* The angles of attack were corrected for wind tunnel wall effect by the standard Prandtl corrections.

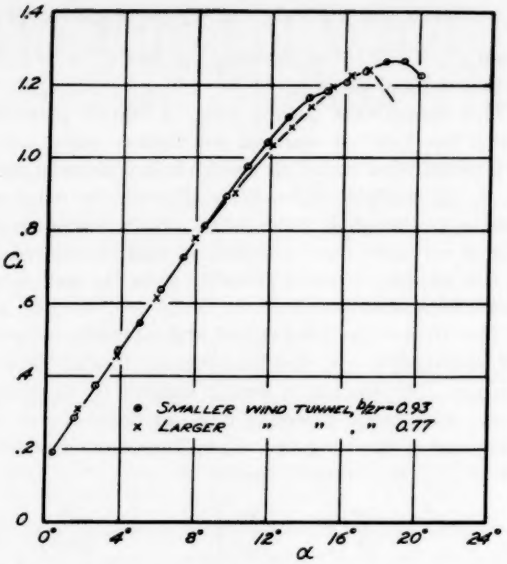


FIG. 4. Lift curves for a wing in two different wind tunnels.

further increased, the entire wing stalled almost simultaneously. In the smaller wind tunnel there were no stalled areas below an angle of attack of 15.1°. At this angle the region from the middle of the aileron to the wing tip stalled suddenly. The stall then progressed gradually inboard and the maximum lift was obtained at $\alpha = 18.2°$. The entirely different type of stall in the two wind tunnels is also shown by the force measurements.

From Fig. 3 it would be predicted that the wing would behave as though it had about 1.9° more washin near the stall in the smaller tunnel. The wing tip was observed to stall in the smaller wind tunnel at an angle of attack 2.2° less than the stalling angle of the tip in the large wind tunnel. The difference in the stalling characteristics in the two wind tunnels is thus almost completely explained by the difference in the induced washin in the two wind tunnels.

From his consideration of the wind tunnel wall corrections to the drag and angle of attack, Millikan concluded that the effect of the distortion of the lift distribution by the wind tunnel walls was not important for the drag and angle of attack corrections as long as $b/2r < 0.95$. From Fig. 3 it is apparent that the effect of this distortion on the stall is noticeable for much smaller values of $b/2r$. For the wing of aspect ratio of 7 at $C_L = 1.2$, the induced washin is 0.5° for $b/2r = 0.8$. This is about the minimum amount of twist that can produce noticeable results in the stall characteristics of a wing; so it may be concluded that the effect of the wind tunnel wall interference on the stalling characteristics of a wing is negligible if $b/2r < 0.8$.

If a wing having a span larger than this is tested in a closed wind tunnel, the induced washin will cause the tip to stall noticeably sooner than it would in free air. A wing which has satisfactory stall characteristics in a closed wind tunnel will thus certainly be satisfactory in free air as the induced washin gives a margin of safety, and tests in a closed section wind tunnel are conservative. On the other hand, the sign of the correction is reversed for a wind tunnel having a circular open working section, and the distortion of the lift distribution tends to prevent tip stall. It is thus possible for a wing to show good stalling characteristics in a wind tunnel with an open jet and still exhibit tip-stall in free air. For this reason it is recommended that considerable caution be used in interpreting wing-stall data obtained in open section wind tunnels unless $b/2r < 0.8$.

REFERENCES

[1] Nazir, P. P., *Tapered Wing Stalling*, Journal of the Royal Aeronautical Society, Vol. XXXIX, No. 296, page 750, August, 1935.

[2] Millikan, Clark B., *On the Stalling of Highly Tapered Wings*, Journal of the Aeronautical Sciences, Vol. 3, No. 5, page 145, March, 1936.

[3] Millikan, Clark B., *On the Lift Distribution for a Wing of Arbitrary Plan Form in a Circular Wind Tunnel*, A.S.M.E. Transactions, Applied Mechanics Section, 1932–1933.

[4] Lotz, I., *Berechnung der Auftriebsverteilung beliebig geformter Flügel*, Z.F.M., Vol. 22, No. 7, page 189, April 14, 1931.

The Aeronautical Index

In January 1935, the Works Division of the Emergency Relief Bureau of the City of New York, known now as the Federal Works Agency, Work Projects Administration, authorized the preparation of an index of aeronautics by specialists provided by the W.P.A. The project was sponsored locally by the Department of Docks of the City of New York.

On June 30, 1941, the project was discontinued as the work that had been authorized had been completed.

It will be of interest to members of the Institute to learn the extent of the files which have been made available to the Institute by the work of the past six years. The appreciation of the Institute is due the officials of the W.P.A. for their cooperative assistance and also to Mayor LaGuardia of New York who sponsored the program.

The Institute provided the material from which the indexes were made, but especial appreciation should be given to the Library of Congress, the New York Public Library and the Engineering Societies Library for permitting research workers to use their facilities.

Card Indexes

The first index prepared was a subject index of aeronautics. When these cards had been compiled, it was decided to publish them in book form so that they would be available to universities, laboratories and government departments. The extent of this work is given in the analysis presented.* It shows that over 23,000 volumes were printed and distributed. These bibliographies and their supplements cover practically the whole range of aeronautics, and from the letters of appreciation received, it is known that they have been of great service. As only two hundred sets could be made, and as they could not be sold, they were distributed over as wide a geographical area as possible, particularly to centers where they would be of the greatest use.

As a by-product of the subject index, a large file of cards was secured which cross-indexed the subjects by authors. In addition, several special indexes were prepared. A chronology of aeronautics was made with the name of each person participating in any event, cross-indexed in the index of persons. A file of English, French, German and American abstracts was made with the cooperation of many organizations. These were also cross-indexed so that they would be readily located. A title file of aeronautical books was made and cross-indexed so that the principal data regarding any aeronautical book could be found.

These indexes now amount to about 2,000,000 references. Sherman M. Fairchild provided the file cabinets to hold this great file collection.

Aeronautical Archives

Before the card index was completed, it was decided by the Council of the Institute to establish the Aeronautical Archives. The growth of this activity of the Institute has had a great expansion during the past two years. W. A. M. Burden placed his collection of 12,000 aeronautical books in the custody of the Archives. The Paul Kollsman Library was established with a grant of $50,000. Mrs. Bella C. Landauer presented her great collection of aeronautical prints, music, trade cards and book plates to the Archives, and Harry Guggenheim added his fine collection of aeronautical art.

Many persons gave collections of clippings, old and current. These have all been filed by names and subjects so that the Institute now has what is undoubtedly the most complete collection of clippings in this country.

The library facilities of the Institute now consist of over 17,000 books, thousands of pamphlets, reprints and photographs. Gifts of valuable books have been received from C. D. Hanscom, Grover Loening, and many others.

A collection of models, which now numbers about one hundred, is being added to by gifts and will in time be one of the great model collections of the world.

Another collection is composed of medals, medallions and insignia. Many of these are on loan, but it represents one of the best collections of its kind. Other collections are of calendars, Christmas cards, posters, and relics of aeronautical events.

All of this valuable material has been the outgrowth of the original work provided by the workers of the Work Projects Administration. The Council of the Institute has authorized the following resolution:

Resolved, that the Institute of the Aeronautical Sciences expresses to the Work Projects Administration and to Mayor La Guardia, of New York City, the appreciation of its membership for the assistance given in compiling an extensive index of Aeronautics from which bibliographies of great value have been provided to universities, libraries, and government departments. This work laid the foundation for the Aeronautical Archives of the Institute and stimulated the acquisition through gifts of a great collection of aeronautical books, prints, and other valuable reference material.

* See following page.

BIBLIOGRAPHIES PREPARED BY THE WORK PROJECTS ADMINISTRATION FROM THE AERONAUTICAL INDEX FILES OF THE INSTITUTE OF THE AERONAUTICAL SCIENCES

		Original Bibliographies						Bibliography Supplements					
		Number of	Number of References			Date Published	Number of	Number of	Number of References			Date Published	Number of
Part	Subject	Pages	English	Foreign	Total		Volumes	Pages	English	Foreign	Total		Volumes
1	Transportation	266	3103	827	3930	1938	265	137	1675	289	1964	1940	502
2	Meteorology	645	5468	4405	9873	1938	260	100	912	261	1173	1940	525
3	Insurance	159	1818	366	2184	1938	255	21	209	5	214	1940	504
4	Dynamics of the Airplane	124	1016	304	1320	1937	260	65	492	212	704	1941	506
5	Seaplanes	86	646	528	1174	1938	245	17	106	93	199	1940	508
6	Flying Boats	94	937	294	1231	1938	245	22	150	70	220	1940	508
7	Amphibians	20	327	51	378	1938	245	8	76	33	109	1940	508
8	Autogiros	34	317	156	473	1938	240	15	125	43	168	1940	504
9	Helicopters	49	434	336	770	1938	240	16	107	75	182	1940	504
10	Gyroplanes & Cyclogiros	9	50	20	70	1938	240	4	33	17	50	1940	504
11	Medicine	117	635	721	1356	1938	600	99	471	611	1082	1940	1000
12	Landing Gears	68	596	246	842	1938	240	26	243	82	325	1940	508
13	Refueling in Flight	5	52	2	54	1938	248	2	27	4	31	1940	508
14	Tailless Airplanes	5	58	35	93	1938	248	1	6	5	11	1940	508
15	Airplane Catapults	9	67	31	98	1938	248	2	15	14	29	1940	508
16	Airplane Carriers	19	157	41	198	1938	248	4	22	14	36	1940	508
17	Diesel Aircraft Engines	74	492	216	708	1937	230	55	464	169	633	1941	543
18	Laws & Regulations	68	306	475	781	1938	240						
19	Control Surfaces	71	652	104	756	1938	230						
20	Slots & Flaps	38	306	75	381	1938	230						
21	Blind Flight	50	494	165	659	1937	230	46	496	151	647	1941	502
22	Radio	120	1142	352	1494	1937	255						
23	Airships	404	3342	2468	5810	1938	230						
24	Air Mail	151	1369	321	1690	1938	250	41	516	29	545	1940	508
25	Air Navigation	64	498	324	822	1939	250						
26	Flight Instruments	92	672	439	1111	1939	250						
27	Propellers	161	1290	480	1770	1937	230						
28	Fuels	201	1431	755	2186	1937	230	168	1497	364	1861	1941	515
29	Lubricants	101	930	234	1164	1937	230	53	395	165	560	1941	508
30	Aerial Photography	210	1641	1257	2898	1939	296						
31	Metal Construction of Aircraft	70	706	152	858	1937	230						
32	Engines—Vol. I	204	1957	627	2584	1940	515						
33	Engines—Vol. II	233	2172	795	2967	1940	515						
34	Engines by Manufacturer	287	2778	985	3763	1940	470						
35	Engine Parts & Accessories	187	2142	714	2856	1939	442						
36	Engine Instruments	50	697	23	720	1939	442						
37	Airports	247	2338	763	3101	1937	430	96	925	317	1242	1941	710
38	Skin Friction & Boundary Flow	98	794	428	1222	1939	250						
39	Stress Analysis	223	1722	799	2521	1939	275						
40	Helium	48	252	136	388	1937	230						
41	Comfort in Aircraft	64	552	135	687	1939	265						
42	Plastic Materials	27	175	42	217	1940	530						
43	Metals & Light Alloys	190	1784	1170	2954	1940	530						
44	Airways	70	560	182	742	1937	230						
45	Wind Tunnels & Laboratories	124	1088	327	1415	1939	510						
46	Gliding and Soaring	140	1032	1182	2214	1937	230						
47	Women in Aeronautics	35	345	85	430	1937	230						
48	Parachutes	70	447	401	848	1938	240	16	121	73	194	1941	508
49	Rocket Propulsion	31	150	202	352	1937	230	4	35	21	56	1941	508
50	Stratospheric Flight	39	284	102	386	1937	230	30	250	133	383	1941	508
	Libr. of Congress Index—1938	325	2716	1254	3970	1940	503						
	Bibliography Index Headings							53	1200	...	1200	1941	504
	Lib. of Congress Index Headings							106	2000	...	2000	1940	1000
	Jeffries—Narrative of the Two Aerial Voyages—1786							88				1941	950
	Totals	6276	54,937	26,532	81,469		15,235	1293	12,568	3250	15,818		15,877

Institute Notes

Hunsaker on Leave as Editor of the Journal

Dr. J. C. Hunsaker, Editor of the Journal of the Aeronautical Sciences since it began publication in January, 1934, will take leave from active editorial supervision of the Journal during the national defense emergency. As announced in these Notes last month, he has been appointed by the Secretary of the Navy to be Coordinator of Research and Development for the Navy Department. On July 31, he was elected Chairman of the National Advisory Committee for Aeronautics.

As Navy Coordinator of Research and Development his duties will be: (1) to advise the Secretary of the Navy on matters of naval research and development; (2) to provide information to bureaus and offices of the Navy regarding research of outside agencies; (3) to cooperate with all agencies of research and development with a view to coordination of effort; (4) to arrange for suitable representation of the Navy on outside boards, committees and councils dealing with research; (5) to provide a program section and a planning section for formulation of coordinated programs of research. In addition he will serve as chairman of the newly established Naval Research and Development Board which will be composed also of representatives of the Chief of Naval Operations, Bureau of Ships, Bureau of Ordnance, Bureau of Aeronautics and Bureau of Yards and Docks.

Dr. Hunsaker has been a civilian member of the National Advisory Committee for Aeronautics since 1938 and was previously a Naval member in 1922–23. As Chairman of the N.A.C.A., he succeeds Dr. Vannevar Bush, who was appointed by President Roosevelt as Director of the Office of Research and Development. Dr. Hunsaker is also the Navy member of the Council of this newly created government office.

Dr. Hunsaker is a graduate of the United States Naval Academy, class of 1908, he was also graduated from Massachusetts Institute of Technology in 1912 and four years later received his degree of Doctor of Science. As an officer of the Naval Construction Corps from 1909 to 1926, he designed the NC-4, first flying boat to cross the Atlantic, and the airship "Shenandoah." He served on the Inter-Allied Naval Armistice Commission in 1918 and as Assistant Naval Attaché to London, Paris, Berlin and Rome from 1923 to 1926. He was awarded the Navy Cross with the following citation:

> "For exceptionally meritorious service in a duty of great responsibility in charge of the Aircraft Division of the Bureau of Construction and Repair, in which position he showed ability unsurpassed in the United States as an aircraft designer as well as great organizing and executive ability in expanding and handling a large force. The successful development of naval aircraft was due largely to the ability and industry of this officer."

Dr. Hunsaker was a Lieutenant Commander when he resigned from the Navy in 1926 to become Vice-President of the Bell Telephone Laboratories. He was Vice-President of the Goodyear-Zeppelin Corporation from 1928 to 1933 and since then has been Head of the Department of Mechanical Engineering and Professor in charge of the Guggenheim Aeronautical Laboratory at M.I.T. Dr. Hunsaker holds a commission as Captain in the U.S. Naval Reserve but is not to be called to active duty as such.

He is a recipient of the Daniel Guggenheim Medal (1933) for achievement in aeronautics, a member of the National Academy of Sciences and the American Society of Naval Architects, a Fellow of the American Academy of Arts and Sciences and the American Physical Society, an Honorary Fellow of the Royal Aeronautical Society of Great Britain and an Honorary Fellow of the Institute of the Aeronautical Sciences, of which he was the first President.

Hugh L. Dryden Appointed Editor

Dr. Hugh L. Dryden, Hon.F.I.Ae.S., Chief of the Mechanics and Sound Division of the national Bureau of Standards, has been appointed, with the approval of the Council, to succeed Dr. J. C. Hunsaker in the editorial supervision of the Journal. During Dr. Hunsaker's leave from editorial duties for the current emergency, Dr. Dryden will assume the responsibility of editing the technical papers in the Journal and acting as Chairman of the Editorial Committee.

Dr. Dryden's experience and eminence in the aeronautical sciences will insure the maintenance of the high editorial and technical standards of the Journal. His activity in aeronautics began upon his graduation from Johns Hopkins University in 1918, from which he received the degrees of M.A. and Ph.D., when he joined the Aerodynamical Physics Section of the National Bureau of Standards. He was made Chief of this Section in 1920 and in 1934 became Chief of the Mechanics and Sound Division.

Dr. Dryden's work has been principally concerned with studies of turbulence in wind tunnels, wind pressure on structures, the characteristics of airfoils at high velocities and measurements of the boundary layer. He directs the work done at the Bureau of Standards in aerodynamics, aircraft structures and aircraft instruments. His many scientific reports have been published as Scientific Papers of the Bureau, Technical Reports of the N.A.C.A. and articles in the Journal of the Aeronautical Sciences and other publications.

On December 17, 1938, he delivered the second Wright Brothers Lecture before the Institute on the subject "Turbulence and the Boundary Layer" in which he presented some results from the minute turbulence measurements made possible by the hot-wire anemometer developed at the Bureau under his supervision. He received the 1940 Sylvanus Albert Reed Award for his notable contributions to the aeronautical sciences made through studies on the mechanics of boundary layer flow and the interpretation of wind tunnel experiments.

A Founder Member of the Institute, he was elected an Honorary Fellow in 1939. He is an Associate Fellow of the Royal Aeronautical Society of Great Britain, a Fellow of the American Physical Society and a member of the Washington Academy of Sciences, the American Association for the Advancement of Sciences and the National Aeronautic Association.

Tenth Annual Meeting and Honors Night Dinner, January 27–30, 1942

The Tenth Annual Meeting of the Institute will be held at Columbia University, New York City, Wednesday, Thursday and Friday, January 28–30, 1942. Morning, afternoon and evening sessions will be held on Aerodynamics, Airplane Design, Air Transport, Materials, Meteorology, Physiology, Power Plants, Radio and Instruments, Rotating Wing Aircraft, and Structures. The sessions on meteorology will be held as joint meetings of the Institute and the American Meteorological Society.

Members wishing to present or sponsor papers for any of the technical sessions are invited to write to the Institute giving a summary or outline of the paper intended for presentation.

The Annual Business Meeting of Members will be on the afternoon of January 28, the last Wednesday in January, as specified in the Constitution.

The Honors Night Dinner will be held at the Waldorf-Astoria in New York on Tuesday, January 27, the evening preceding the technical sessions of the Annual Meeting. It will mark the beginning of the Institute's tenth year of activity as a scientific aeronautical society.

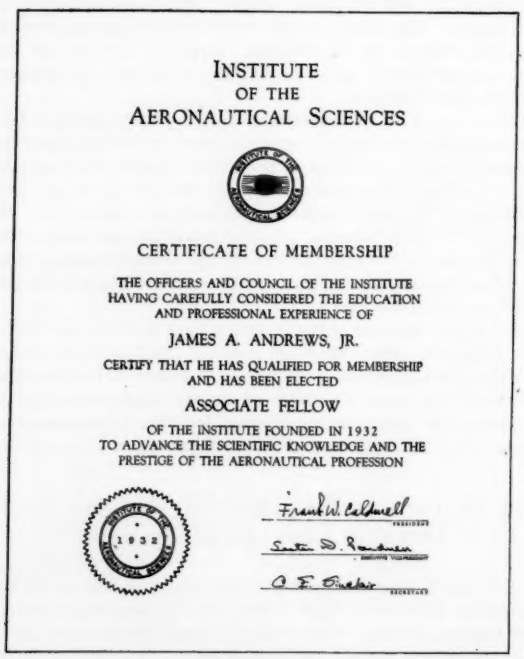
President Caldwell Appointed Representative on National Research Council

President Frank W. Caldwell has been appointed to represent the Institute on the Division of Engineering and Industrial Research of the National Research Council, it is announced by Dr. W. F. Durand, Chairman of the Division. The appointment was made by Dr. F. B. Jewett, President of the National Academy of Sciences, the Academy being the parent body which established the National Research Council.

Mr. Caldwell's appointment is for a term of three years from July 1, 1941. He fills the vacancy left by Elmer A. Sperry, Jr., previously representing the Institute, whose term expired on that date.

New Certificate of Membership

At its last meeting, the Council of the Institute approved the design for a new certificate of membership, a reproduction of which is shown in the accompanying illustration.



INSTITUTE
OF THE
AERONAUTICAL SCIENCES

CERTIFICATE OF MEMBERSHIP

THE OFFICERS AND COUNCIL OF THE INSTITUTE
HAVING CAREFULLY CONSIDERED THE EDUCATION
AND PROFESSIONAL EXPERIENCE OF

JAMES A. ANDREWS, JR.

CERTIFY THAT HE HAS QUALIFIED FOR MEMBERSHIP
AND HAS BEEN ELECTED

ASSOCIATE FELLOW

OF THE INSTITUTE FOUNDED IN 1932
TO ADVANCE THE SCIENTIFIC KNOWLEDGE AND THE
PRESTIGE OF THE AERONAUTICAL PROFESSION

1932

Frank W. Caldwell
PRESIDENT

Secretary

Director

The new certificate is printed in black on heavy white "ledger" paper, 11 inches wide by 14 inches in length. Each certificate will have the name and grade of the member printed in to match exactly the rest of the text, will be signed by three officers of the Institute and will have the corporation seal of the Institute embossed in gold at the lower left corner.

The Officers and Council have been considering for some time the desirability of replacing with a new design the membership certificate printed on blue paper which has been used for the past several years. It is felt that the design just adopted is a distinct improvement over the old, providing a more suitable type of certificate for membership in a scientific society and one which members will find more decorative and appropriate for framing.

The certificates will be available in October at $1.00 each, which covers all handling and mailing charges. An order blank to be used by members who wish to replace their old certificates with the new one is sent to all members with their bills for annual dues.

Kahn Replaces de Florez on Council

At its last meeting the Council of the Institute accepted with regret the resignation of Commander Luis de Florez, U.S.N., from membership on the Council.

An oil technologist and private pilot, Commander de Florez has been active in the affairs of the Institute since its establishment. However, because of his present tour of duty at the Navy Bureau of Aeronautics he felt unable to devote sufficient time to the work of the Institute Council.

By unanimous vote, Roger Wolfe Kahn was elected to fill this vacancy on the Council until the next Annual Meeting. A sportsman pilot well known for his activities in private flying, Mr. Kahn has been a member of the Institute since 1935. He has done test flying and was one of the first pilots to demonstrate an air mail pick-up device.

News of Institute Members

President Roosevelt has appointed **Dr. William F. Durand,** Hon.F.I.Ae.S., to membership on the National Advisory Committee for Aeronautics, succeeding Dr. Robert E. Doherty, President of Carnegie Institute of Technology, who recently resigned. Dr. Durand, Emeritus Professor of Mechanical Engineering of Stanford University, was one of the original members of the N.A.C.A. appointed by President Wilson in 1915. He served as Chairman of the Committee in 1917 and 1918, retiring from membership in 1933.

Aaron Bloom, Technical Member, Assistant Instructor in airplane structures at the Air Corps Technical School, Urbana, Illinois, has been transferred to the new Air Corps school at Sheppard Field, Texas, to assist in organizing similar courses there.

Col. Henry Breckinridge, M.I.Ae.S., has been elected Chairman of the Board of Dowty Equipment Corp.

Gardner W. Carr, M.I.Ae.S., previously with Lockheed Aircraft Corp. as supervisor of materials and later coordinater of British contracts for Lockheed and Vega, is now Executive Vice-President of McDonnell Aircraft Corp.

Howard Carson, Technical Member, has recently become an Assistant Project Engineer in the Aeroproducts Division, General Motors Corp., at Dayton.

T. A. Cheatham, Jr., Technical Member, is now a Junior Aeronautical Engineer, Bureau of Aeronautics, Navy Department in Washington. He was previously Aeronautical Engineering Aide at the Naval Aircraft Factory, Philadelphia.

G. Graham Davidson, M.I.Ae.S., formerly Aeronautical Engineer at the St. Louis Plant of Curtiss-Wright Corporation's Airplane Division, is Structures Engineer at the Columbus, Ohio, plant.

Philip J. Grande, Technical Member, on the staff of Lockheed Aircraft Corp., has been transferred from sheet metal fabrication to the position of Junior Detail Designer on Armaments.

A. H. Hall, F.I.Ae.S., who retired in July from the post of Chief Superintendent of the Royal Aircraft Establishment at Farnborough, England, is succeeded by **W. S. Farren,** F.I.Ae.S., who was until recently, Director of Technical Development in the British Ministry of Aircraft Production.

Brigadier General Henry W. Harms, A.F.I.Ae.S., formerly Commandant of the West Coast Air Corps Training Center at Moffet Field, Calif., has been transferred to the post of Commanding General of the Newfoundland Base Command.

George T. Hayes, Technical Member, on active duty as an Ensign in the U.S. Naval Reserve, is Assistant Inspection and Test Officer, A. & R. Department, at the Pensacola Naval Air Station.

Robert L. Jahnke, Technical Member, has taken a position as Test Engineer with the Allison Division of General Motors Corp.

Neil Lamont, Jr., Technical Member, has left the Douglas Aircraft Co. where he was a Weight Engineer, to go on active duty as 2nd Lieutenant in the Air Corps. He is stationed at the Basic Flying School at San Angelo, Texas.

Major Reed G. Landis, M.I.Ae.S., Vice-President of American Airlines, Inc., has been named by Mayor F. H. La Guardia, Director of Civilian Defense, to serve as Regional Defense Director for the Sixth Corps Area, Chicago.

O. T. Larson, M.I.Ae.S., has been promoted from the position of General Superintendent of Trans-Canada Airlines, which he held since September, 1939, to Vice-President of Trans-Canada, succeeding D. B. Colyer who resigned from that office on August 20.

Richard J. Markey, M.I.Ae.S., until recently in charge of final assembly planning at the plant of Vought-Sikorsky Aircraft Division, United Aircraft Corp., is now representative for Vought-Sikorsky at Intercontinent Aircraft Corp. in Miami, Florida.

P. P. Nazir, M.I.Ae.S., is now with the Inspection Department of National Steel Car Corp., Aircraft Division, at Walton, Ontario.

J. Story Smith, M.I.Ae.S., has resigned from the presidency of Jacobs Aircraft Engine Company to become a Vice-President and to take over the office of Secretary-Treasurer relinquished by H. Alvin Cressman. **C. J. Abbott,** M.I.Ae.S., formerly Vice-President and General Manager of the Jacobs company was elected President.

J. Parker Van Zandt, A.F.I.Ae.S., consulting economic analyst for the Civil Aeronautics Board left in August for a month's survey trip in South America where he will study the air transportation systems of Peru and Ecuador.

R. O. Wickersham, Technical Member, who was Assistant Professor of Mechanical Engineering at the University of Maryland during the past school year, has joined the Department of Mechanical Engineering of Pennsylvania State College as Assistant Professor of Aeronautical Engineering.

Walter C. Williams, Technical Member, formerly with The Glenn L. Martin Co., has joined the staff of the N.A.C.A. as a Junior Engineer, Flight Research Section at the Langley Memorial Aeronautical Laboratories.

Dr. John E. Younger, F.I.Ae.S. and Member of the Advisory Board, received the Spirit of St. Louis Medal, endowed by citizens of St. Louis, at the Semi-Annual Meeting of the American Society of Mechanical Engineers in Kansas City, Mo., in June. The medal is awarded by the A.S.M.E. approximately every three years "for meritorious service in the advancement of aeronautics." Dr. Younger, who is Professor and Head of the Department of Mechanical Engineering at the University of Maryland, has made notable contributions to airplane design, particularly in relation to the pressure-cabin type of high altitude airplane.

Sections

Seattle. Gaylord W. Newton of the Boeing Aircraft Company presented a paper on "Unusual Power Plant Accidents in Civil Aviation," at a meeting held on May 21. The following officers were elected: M. L. Pennell, Chairman; R. E. Hage, Vice-Chairman; Robert Plath, Secretary; and J. P. Butler, Treasurer.

Student Branches

University of Cincinnati. Officers elected at a meeting held on April 9, are as follows: H. P. Liepmann, Honorary Chairman; Robert Tripp, Chairman; Monro Duke, Vice-Chairman; Thomas Tripp, Secretary Section I; Bruce Hartman, Secretary Section II. Talks by James Kordenbrock and Wayne Bryant on "Modern Trends in Aircraft Design" and "Plastics" were presented, respectively.

Virginia Polytechnic Institute. The following officers were elected at a recent meeting: John H. Wright, Honorary Chairman; J. W. Martin, Jr., Chairman; E. F. Burner, Vice-Chairman.

New Corporate Members

The following companies became Corporate Members of the Institute last month:

Breeze Corporations, Inc., Newark, N. J., accessory manufacturers and fabricators of stainless steel. The company manufactures radio ignition shielding, flexible metal conduit and fittings, and the cartridge type engine starter, as well as instruments, controls and armor plate for aircraft and light tanks.

Warner Aircraft Corporation, Detroit, Michigan, makers of the "Scarab" line of light radial air-cooled engines for aircraft and manufacturers of airplane wheels and brakes, shock struts and other aircraft accessories.

Personnel Opportunities

The Personnel Bureau serves individual members, as well as organizations seeking to employ aeronautical specialists. Any member or organization may have requirements listed without charge.

Wanted

Aircraft Production Executive. A capable aircraft production executive qualified by a knowledge of aircraft construction and experience in aircraft factory methods. Must have the executive ability to organize and direct the activities of approximately five thousand employees in a large, well established corporation. Unusual opportunity and salary to correspond with the responsibility involved for the qualified applicant. Only those stating age, citizenship status, abstract of experience, and salaries earned will be considered. Address reply to Box 138, Institute of the Aeronautical Sciences.

Instructors. Engineering school in midwest expects to have openings in the near future for instructors qualified to teach any of the following subjects: Physics, Strength of Materials, Calculus, Aerodynamics, Stress Analysis and Airplane Design. State experience and salary desired. Address reply to Box 139, Institute of the Aeronautical Sciences.

Advertising Executive. Nationally known Ohio manufacturer now engaged in large-scale production of airplane parts wants experienced man to direct advertising campaign to keep this activity before the public and the trade. Age 28–38. Successful aviation advertising experience desirable. Reply should state full qualifications and salary expectancy. Address reply to Box 140, Institute of the Aeronautical Sciences.

Changes in Membership

The following changes in the membership of the Institute have occurred since the publication of the previous issue of the Journal.

Transferred to Associate Fellow Grade

Bollay, William, Ph.D. in Ae.E.; Instructor in Applied Mechanics, Graduate School of Engineering, Harvard University. (See A.M. of S.)

Bryan, Guy Lee, Jr., M.C.E.; Chief Structural Engineer, The Glenn L. Martin Co.

DuBois, Ralph Nicholas, B.S.E. in M.E.; Executive Engineer, Aircraft Engine Div., Packard Motor Car Co. (See W.W. in Eng.; W.W. in Aviation.)

Elected to MEMBER Grade

Anderson, Newton H., B.S. in M.E.; Engineer, Program Director, Education Dept., Douglas Aircraft Co.

Braddon, Frederick Darcy, M.E.; Research Engineer, Sperry Gyroscope Co.

Burke, John Francis, B.Ae.E.; Meteorologist, Pan American Airways System (Miami).

Franklin, J. C., M.B.A.; Superintendent of Engineering, Transcontinental and Western Air, Inc.

Hufferd, George H., Chief Engineer, Detroit Plant, Thompson Products, Inc.

Jewett, Frederic Davis, Ae.E.; Project Stress Engineer, The Glenn L. Martin Co.

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